N73-20018

NASA CR-112250
March 1973

FLIGHT SERVICE EVALUATION
OF PRD-49/EPOXY COMPOSITE PANELS
IN WIDE-BODIED COMMERCIAL TRANSPORT AIRCRAFT

Final Report

by

John H. Wooley, Dale R. Paschal and Eugene R. Crilly

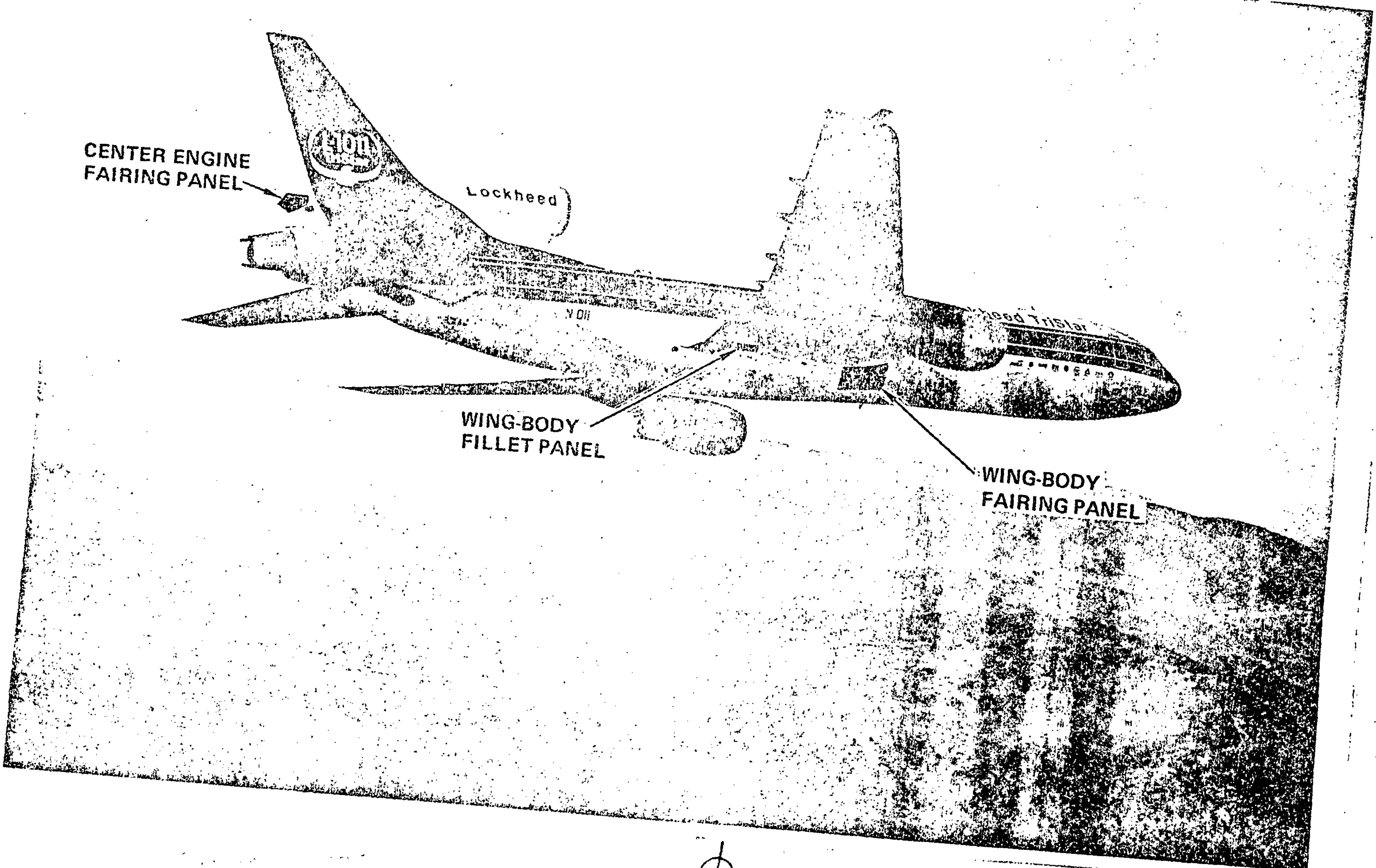
Prepared under Contract No. NAS1-11621

For

Langley Research Center
National Aeronautics and Space Administration
Hampton, Virginia 23365

by

Lockheed-California Company
Burbank, California
A Division of Lockheed Aircraft Corporation


CENTER ENGINE
FAIRING PANEL
Lockheed
WING-BODY
FILLET PANEL
WING-BODY
FAIRING PANEL

FOREWORD

This technical report was prepared by the Lockheed-California Company under Contract No. NAS1-11621, "Flight Service Evaluation of PRD-49 Composite Panels in Wide Bodied Commercial Transport Aircraft". It summarizes the work performed during the period of May through November 1972 which includes testing, machining development, fabrication, manufacturing cost studies and installation of PRD-49 fairing panels. The remainder of the program involves a five year evaluation of the panels in commercial airline service.

This program has been administered by the Langley Research Center National Aeronautics and Space Administration with Mr. Benson Dexter of the Materials Division being the Project Engineer.

Individuals who have made contributions at the Lockheed-California Company and Heath Tecna Corp., the fabricator of the panels, are as follows:

John H. Wooley	Program Leader
Dale R. Paschal	Commercial Engineering, L-1011 Project Coordinator
Eugene R. Crilly	Science and Engineering Coordinator
Lloyd W. Nelson	Commercial Engineering Structures Division Engineer
Donald J. Mackey	Commercial Engineering, Stress Department Engineer
Owen M. Klasen	Structures Laboratory Engineer
J. S. Southworth	Structures Engineer, Sr.
John VanHamersveld	Science and Engineering Staff Engineer
I.L. Bertrand	Manufacturing Research Engineer
Glen H. Smith	Manufacturing Research Engineer
I.L. Smith	L-1011 Assembly Department Manager
T. Johnson	L-1011 Assembly Supervisor
Donald Colclough	Heath Tecna Manufacturing Engineering

TABLE OF CONTENTS

TABLES

FIGURES

LIST OF SYMBOLS AND ABBREVIATIONS

Symbol	Description
FAA	Federal Aviation Administration
hr	hour
kg	kilogram
kg/m^2	kilograms per square meter
kg/m^3	kilograms per cubic meter
ksi	one thousand pound force per square inch
lb/cu.in.	pounds per cubic inch
m	meter
mm	millimeter
NDT	Non-Destructive Testing
N/m^2	Newtons per square meter
oz/sq.yd.	ounces per square yard
P/N	Part Number
psi	pounds force per square inch
RT	Room Temperature
TPI	Turns per Inch
°F	Temperature - Degrees Fahrenheit
°K	Temperature - Degrees Kelvin
$\$/m^2$	dollars per square meter
$/sq.yd.	dollars per square yard

SUMMARY

Three L-1011 fairing panel configurations were selected as test parts to compare the fabrication, costs and service performance characteristics of PRD-49 and fiberglass. These parts are currently fiberglass reinforced structure and the purpose of this program is to evaluate the results of direct substitution of PRD-49 fabric for the fiberglass. Three ship sets of these panels have been fabricated for a five year flight service evaluation on three L-1011 commercial airlines operating in widely diverse route structures.

The same epoxy resin systems were used for the PRD-49 fabric to eliminate matrix variables and maintain the same processing procedures. The simple replacement of fiberglass with PRD-49 in these panels (six per ship set) saved 7.35 kg (16.2 pounds) per aircraft or 26.6 percent of the weight of the fiberglass panels.

The standard tools and machining techniques used for fiberglass parts are unacceptable for cutting, trimming, and drilling the tougher PRD-49 fibers. Therefore, a machining development study was undertaken to provide the necessary new tools and machining techniques. After incorporating these new developments in the fabrication and installation of the panels, a manufacturing cost study revealed that the labor hours were only increased by about 12.5 percent. This results in an added cost of $ 33.00 per kg ($15.00 per pound) of weight saved. Material cost increases amounted to $ 113.00 per kg ($ 51.50 per pound) of weight saved for the large wing-to-body fairing panel, $ 123.00 per kg ($ 56.00 per pound) for the wing-to-body fillet and $ 303.00 per kg ($ 137.50 per pound) for the center engine fairing.

"Page missing from available version"

1.0 INTRODUCTION

The objective of this program is to provide a means of comparing manufacturing techniques, costs and long time commercial airline service performance of DuPonts' new, lightweight PRD-49 fabric with the conventional fiberglass fabric.

Three fiberglass fairing panel configurations on the Lockheed L-1011 were selected as test articles for this evaluation. These panels are described in Table 1-1 and are shown in Figure 1-1.

The L-1011 provides an excellent means of evaluating the service performance of PRD-49 fabric since the various commercial airline customers log up to 3000 flight hours per aircraft each year in widely diverse environments. A set of each of the above mentioned panels (left and right hand sides) will be flight tested for five years on a TWA aircraft having transcontinental flights, an Air Canada aircraft which is exposed to the cold northern climate and an Eastern Air Lines aircraft which operates in the eastern seaboard environment. An estimated 270,000 hours of flight service will be logged by these eighteen panels over the 5-year service evaluation period.

Additional environmental exposure data will be obtained from 200 flexural specimens, 200 compression specimens and 200 interlaminar shear specimens, which have been fabricated with two resin systems, for testing by NASA Langley over the five year period.

Prior to obtaining FAA and airline approval to install this new material on the aircraft, several steps were taken to assure that the structural integrity and reliability of the parts were not jeopardized. Material and Process Specification requirements were established, physical and mechanical properties were determined and two of the large wing-to-body fairing panels were static tested to failure.

Prior to the start of this program, it was recognized that one of the

TABLE 1-1. Description of Test Panels

Part	Part Number	Type of Construction	Max. Dim.	Approx. Surface Area	Maximum Service Temp.
Wing-to-Body Fairing Panel	1515599-109 (LH) 1515599-110 (RH)	Honeycomb Panel- Curved Rectangular	1.52m x 1.70m (60 in x 67 in)	$2.59 m^2$ (27.9 sq.ft)	$344^\circ K$ ($160^\circ F$)
Wing-to-Body Fairing Fillet	1545328-109 (LH) 1545328-110 (RH)	Solid Laminate - Z Shape with sharp radii	.23m x.84m (9 in x 33 in)	$0.19 m^2$ (2.1 sq.ft)	$344^\circ K$ ($160^\circ F$)
Center Engine Fairing Panel	1538592-129 (LH) 1544685-117 (RH)	Honeycomb Panel - Approximately Triangular	.76m x 1.83m (30 in x 72 in)	$1.39 m^2$ (7.0 sq.ft)	$422^\circ K$ ($300^\circ F$)*

* Soakback Condition on Engine Shut-Down

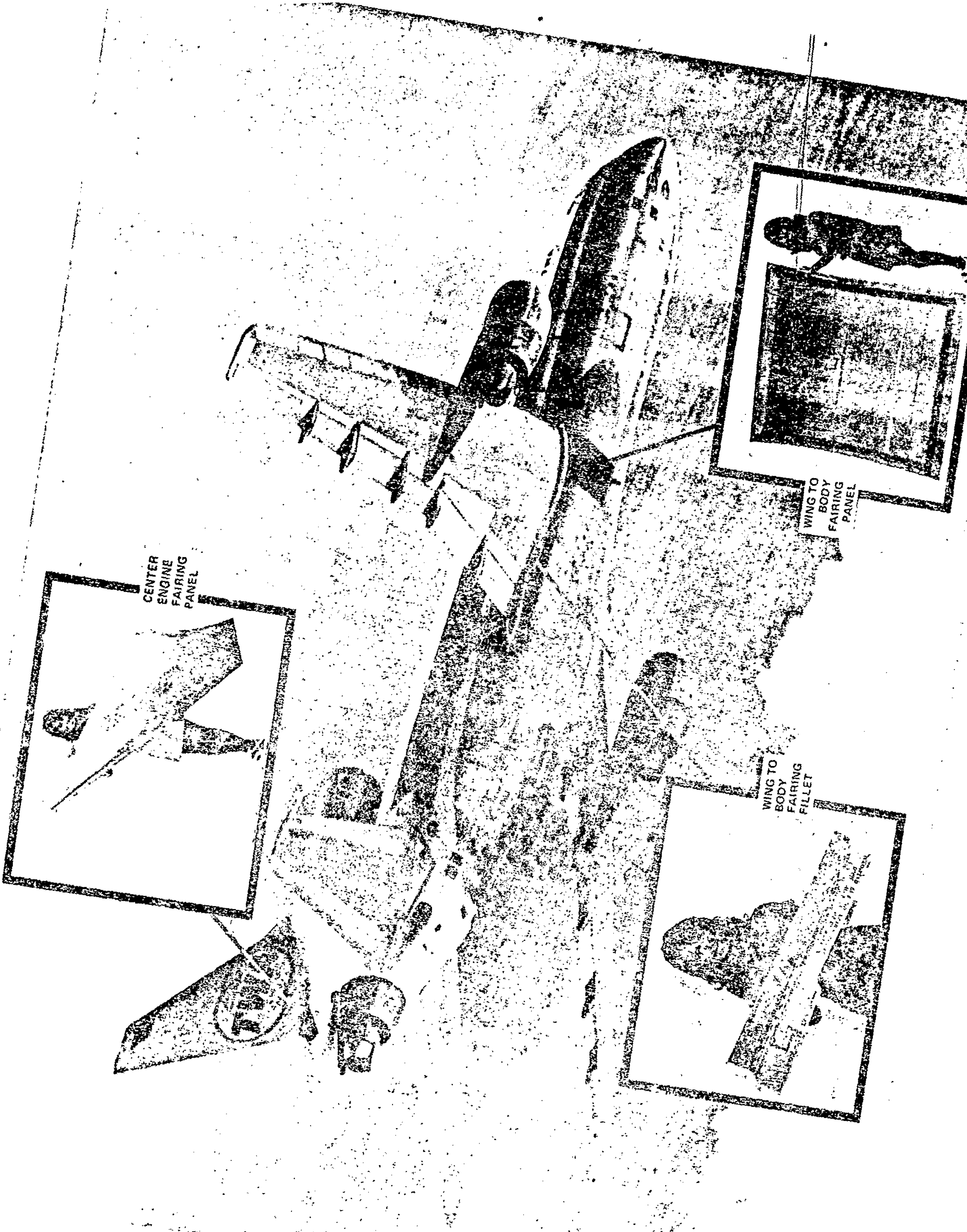


Figure 1-1 PRD-49 Fairings

major problems associated with the use of PRD-49 fabric was the trimming, drilling and countersinking of cured parts. Hence, in addition to fabrication of the parts for flight service evaluation, studies were performed to develop satisfactory methods of accomplishing the above mentioned machining operations. Cost differences between fabrication of PRD-49 and fiberglass parts were also identified. The data thus accumulated should help to point out any differences between the two materials and provide answers to some of the unknowns associated with the use of PRD-49.

2.0 MATERIALS

The low density of PRD-49 makes this fiber very attractive for new composite aircraft applications as well as a replacement for fiberglass on existing aircraft parts. The tough, abrasion resistant characteristics of the PRD-49 fiber permit weaving into fabrics having properties and handling characteristics very similar to comparable styles of fiberglass, yet the fiber density of PRD-49 is 43 percent lower. A 30-35 percent weight savings can be realized by direct substitution in laminates and 18 - 28 percent can be saved in sandwich structure depending on the type and quantity of core used. For this program two PRD-49 fabric styles were used. One was a 0.17 kg/m^2(5 oz./sq. yd), 0.25 mm (0.010 inch) thick weave similar to 181 style fiberglass [8 harness satin 0.30 kg/m^2(8.8oz/sq. yd)] and the other was a 0.06 kg/m^2 (1.8 oz/sq. yd), 0.13 mm (0.005 inch) thick plain weave material comparable to 0.11 kg/m^2 (3.2 oz/sq. yd), 0.13 mm (0.005 inch) thick fiberglass. A comparison of the fiber and fabric characteristics is given in Tables 2-1 and 2-2, respectively. The PRD-49 specification requirements used for controlling the material are indicated in the Tables.

In considering resin systems for use in this program, Hexcel's F-155 epoxy resin was selected for the 344 $^{\circ}$K (160°F) service environment of the wing-to-body honeycomb fairing panels and solid laminate fillets, and their F-161 epoxy resin system was used for the 422°K (300°F) service of the center engine honeycomb fairing panels. Previous tests with other resin systems in combination with PRD-49 had produced comparable results, however, the Hexcel system was selected since it is the resin currently used on the L-1011 fiberglass fairings. This permitted a direct comparison of PRD-49 with fiberglass by eliminating many other variables inherent in the use of different resin systems. It also permitted the use of the same tooling, bagging techniques and cure cycles during fabrication of the fairing panels.

The impregnated material was ordered to the requirements of a Lockheed-California Company Material Specification. Results of the acceptance testing are provided in Tables2-3 and 2-4 along with the specification requirements. Additional property data obtained from the process control specimens is shown in

TABLE 2-1. COMPARISON OF "E" GLASS AND PRD-49 FIBER PROPERTIES

Material	Nominal Fiber Diameter	Nominal Fiber Tensile Strength	Nominal Fiber Modulus	Nominal Fiber Density
PRD-49 Type III	0.012 mm (0.46 mils)	2.758×10^9 N/m^2 (400 ksi)	124×10^9-138×10^9 N/m^2 (18×10^6-20×10^6 psi)	1.47×10^3 kg/m^3 (.053 lb/cu. in.)
"E" Glass	0.009 mm (0.36 mils)	2.413×10^9 N/m^2 (350 ksi)	72.4×10^9 N/m^2 (10.5×10^6 psi)	2.49×10^3 kg/m^3 (.090 lb/cu.in.)

TABLE 2-2 COMPARISON OF PRD-49 AND FIBERGLASS FABRICS

Material	Fabric Style	Nominal Wt.	Nominal Thickness		Weave	Warp & Fill Yarn	Nominal End Count Yarns/Inch	
			Fabric	Per Ply of Cured Laminate			Warp	Fill
PRD-49 Type III	0.169 kg/m^2 (5 oz/sq.yd)	0.161-0.178 kg/m^2 (4.75 - 5.25 * oz/sq.yd)	0.229 mm (9 mils)	0.254 mm (10 mils)	8 Harness* Satin	380 Denier* .9 TPI	48 $\pm$ 2*	48 $\pm$ 2*
	0.061 kg/m^2 (1.8 oz/sqyd)	0.056-0.064 kg/m^2 (1.66-1.94* oz/sq.yd.)	0.102 mm (4 mils)	0.127 mm (5 mils)	Plain*	195 Denier* .9 TPI	34 $\pm$ 2*	34 $\pm$ 2*
"E" Glass	181	0.271-0.339 kg/m^2 (8.0-10.0 oz/sq.yd.)	0.229 mm (9 mils)	0.254 mm (10 mils)	8 Harness Satin	150 1/2	60	58
	120	0.091-0.114 kg/m^2 (2.68 - 3.35 oz/sq.yd.)	0.102 mm (4 mils)	0.127 mm (5 mils)	Crowfoot	450 1/2	57	54

* Lockheed Specification Requirements

TABLE 2-3 ACCEPTANCE TEST DATA

$344^{o}K$ ($160^{o}F$) Resin System

Property	0.17 kg/m^2 (5 oz/sq.yd.) Fabric		0.061 kg/m^2 (1.8 oz/sq.yd.) Fabric	
	Spec. Requirement	Actual	Spec. Requirement	Actual
Wet Resin Content Percent by Weight	42-48	48.0 47.1	47-53	51.6
Volatiles Percent by Weight	2 Max.	1.1 1.3	2 Max.	1.0
Gel Time Seconds	180-660	285	180-660	300
Tensile Ultimate Dry R.T. N/m^2 (psi)	4.137×10^8 (60,000)	4.295×10^8 (62,300) 4.240×10^8 (61,500) 4.213×10^8 (61,100) 4.151×10^8 (60,200) 4.482×10^8 (65,000) 4.276×10^8 (62,100)	3.792×10^8 (55,000)	3.875×10^8 (56,200) 4.102×10^8 (59,500) 4.137×10^8 (60,000) 3.826×10^8 (55,500) 3.936×10^8 (57,100) 3.975×10^8 (57,600)
Tensile Modulus Dry R.T. N/m^2 (psi)	30.34×10^9 (4.4×10^6)	29.65×10^9 (4.3×10^6) 30.34×10^9 (4.4×10^6) 31.72×10^9 (4.6×10^6) 31.03×10^9 (4.5×10^6) 32.41×10^9 (4.7×10^6) 31.03×10^9 (4.5×10^6)	26.2×10^9 (3.8×10^6)	25.51×10^9 (3.7×10^6) 26.89×10^9 (3.9×10^6) 26.89×10^9 (3.9×10^6) 26.20×10^9 (3.8×10^6) 26.89×10^9 (3.8×10^6) 26.48×10^9 (3.8×10^6)
Compressive Ultimate Wet R.T. N/m^2 (psi)	1.379×10^8 (20,000)	1.551×10^8 (22,500) 1.420×10^8 (20,600) 1.358×10^8 (19,700) 1.482×10^8 (21,500) 1.420×10^8 (20,600) 1.446×10^8 (21,000)	1.379×10^8 (20,000)	1.317×10^8 (19,000) 1.538×10^8 (22,300) 1.420×10^8 (20,600) 1.434×10^8 (20,800) 1.462×10^8 (21,200) 1.434×10^8 (20,800)
Sandwich Flatwise Tensile R.T. N/m^2 (psi)	2.068×10^6 * (300)	** 2.068×10^6 (300) 2.172×10^6 (315) 1.965×10^6 (285) 2.068×10^6 (300) 2.034×10^6 (295) 2.061×10^6 (299)	2.068×10^6 * (300)	** 1.965×10^6 (285) 2.068×10^6 (300) 2.034×10^6 (295) 2.137×10^6 (310) 2.206×10^6 (320) 2.082×10^6 (302)

* Core failure below this value acceptable

** All specimens failed in core

TABLE 2-4 ACCEPTANCE TEST DATA

422^{o}K (300^{o}F) Resin System

Property	0.17 kg/m^2 (5 oz/sq.yd.) Fabric		0.061 kg/m^2 (1.8 oz/sq.yd.) Fabric	
	Spec. Requirement	Actual	Spec. Requirement	Actual
Wet Resin Content Percent by Weight	42-48	46.8	47-53	52.1
Volatiles Percent by Weight	5 Max.	.99	5 Max.	1.1
Gel Time Seconds	60-420	180	60-420	180
Tensile Ultimate Dry, R. T. N/m^2 (psi)	$4.137x10^8$ (60,000)	$4.171x10^8$ (60,500) $4.295x10^8$ (62,300) $4.557x10^8$ (66,100) $4.233x10^8$ (61,400) $4.351x10^8$ (63,100) $4.321x10^8$ (62,600)	$3.792x10^8$ (55,000)	$3.937x10^8$ (57,100) $3.806x10^8$ (55,200) $3.868x10^8$ (56,100) $4.020x10^8$ (58,300) $3.875x10^8$ (56,200) $3.902x10^8$ (56,600)
Tensile Modulus Dry R.T. N/m^2 (psi)	$30.34x10^9$ ($4.4x10^6$)	$33.09x10^9$ ($4.8x10^6$) $29.65x10^9$ ($4.3x10^6$) $31.03x10^9$ ($4.5x10^6$) $31.72x10^9$ ($4.6x10^6$) $31.72x10^9$ ($4.5x10^6$) $31.03x10^9$ ($4.5x10^6$)	$26.20x10^9$ ($3.8x10^6$)	$26.20x10^9$ ($3.8x10^6$) $25.51x10^9$ ($3.7x10^6$) $26.89x10^9$ ($3.9x10^6$) $27.58x10^9$ ($4.0x10^6$) $26.20x10^9$ ($3.8x10^6$) $26.48x10^9$ ($3.8x10^6$)
Compressive Ultimate Wet R. T. N/m^2 (psi)	$1.379x10^8$ (20,000)	$1.400x10^8$ (20,300) $1.475x10^8$ (21,400) $1.365x10^8$ (19,800) $1.558x10^8$ (22,600) $1.544x10^8$ (22,400) $1.468x10^8$ (21,300	$1.379x10^8$ (20,000)	$1.538x10^8$ (22,300) $1.462x10^8$ (21,200) $1.420x10^8$ (20,600) $1.317x10^8$ (19,100) $1.469x10^8$ (21,300) $1.441x10^8$ (20,900)
Sandwich Flatwise Tensile R.T. N/m^2 (psi)	$2.068x10^{6}$* (300)	$2.137x10^6$ (310)** $2.179x10^6$ (316) $2.220x10^6$ (322) $2.206x10^6$ (320) $2.068x10^6$ (300) $2.162x10^6$ (313)	$2.068x10^{6}$* (300)	$2.034x10^6$ (295)** $2.137x10^6$ (310) $2.193x10^6$ (318) $2.206x10^6$ (320) $2.227x10^6$ (323) $2.162x10^6$ (313)

* Core failure below this value acceptable

** All specimens failed in core

Table 4-1 in the Fabrication Section.

The core used in the honeycomb panels was 3.17 mm (1/8 inch) cell size, 48.1 kg/m^3 (3 lb/ft^3) density Nomex(R) (Hexcel's HRH-10) .

(R) Du Pont Registered Trademark

3.0 STRUCTURAL ANALYSIS

In order to prove the feasibility of using PRD-49 in L-1011 wing-to-body fairing panels, the largest of the panels, P/N 1515599, was fabricated and statically tested in the fixture shown in Figures 3-1 and 3-2. This panel has a width of 1.52 meters (60 inches) and a length of about 1.7 meters (67 inches). It is contoured to airplane loft lines. In the original test panel, a ply for ply substitution for fiberglass was made so that the outer skin consisted of two plies of 0.13 mm (0.005 inch) thick PRD-49/epoxy [.06 kg /m^2 (1.8 oz/sq.yd)] in lieu of 120 style fiberglass and one ply of 0.254 mm (0.010 inch) PRD-49 [.17 kg /m^2 (5.0 oz./sq. yd)]in lieu of 181 style fiberglass. The inner skin was made up of three plies of the lighter weight PRD-49 fabric. The edge band was built up for attachment purposes to approximately 2.54 mm (0.100 inch). This panel was subjected to a static internal pressure test of 8.27×10^3 N/m^2 (1.2 psi) and then tested to failure with external pressure. Design ultimate for external pressure is 16.55×10^3 N/m^2 (2.4 psi). In this initial external pressurization test, a crack was noted propagating from a fastener hole at approximately mid-span in the 1.7 meter (67 inch) direction of the panel at about design ultimate of 16.55×10^3 N/m^2 (2.4 psi) but the panel continued to carry load until the pressurization bladder failed at about 20.34×10^3 N/m^2 (2.95 psi). The calculated skin stress was 125.46×10^6 N/m^2 (16,600 psi) when analyzed by the methods given in Appendix A. Lockheed's preliminary design allowable in compression for this material is 121.01×10^6 N/m^2 (16,100 psi) when an 0.8 multiplying factor is used to compensate for the thin material 0.508 mm (0.020 inch). Most test data is obtained on laminates 3.175 mm (0.125 inch) thick.

Since the original panel was still carrying load when the pressurization bladder failed, it was decided to test a new panel with three plies of the light weight fabric on the outer face in an effort to determine if a more efficient structure could be utilized. This panel failed at 14.13×10^3 N/m^2 (2.05 psi) demonstrating that the thinner face sheet, 0.381 mm (0.015), was not structurally acceptable. The failure mode was a compression buckle on the outer (compression) skin in the flat portion of the panel as shown in Figures 3-3 and 3-4. The fiber stress on this face at failure was calculated to be

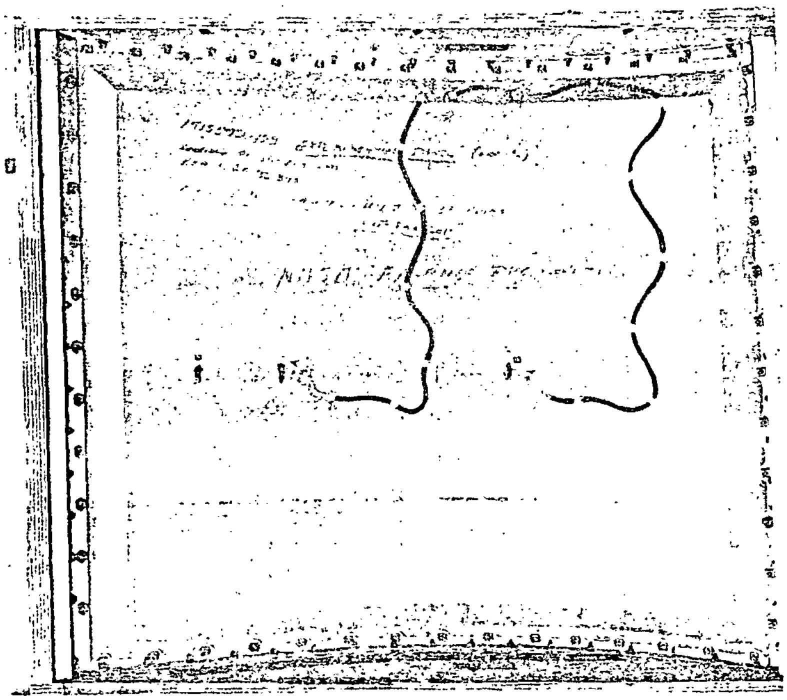


Figure 3-1 Wing-to-Body Fairing Panel in Fixture for External Pressurization Test

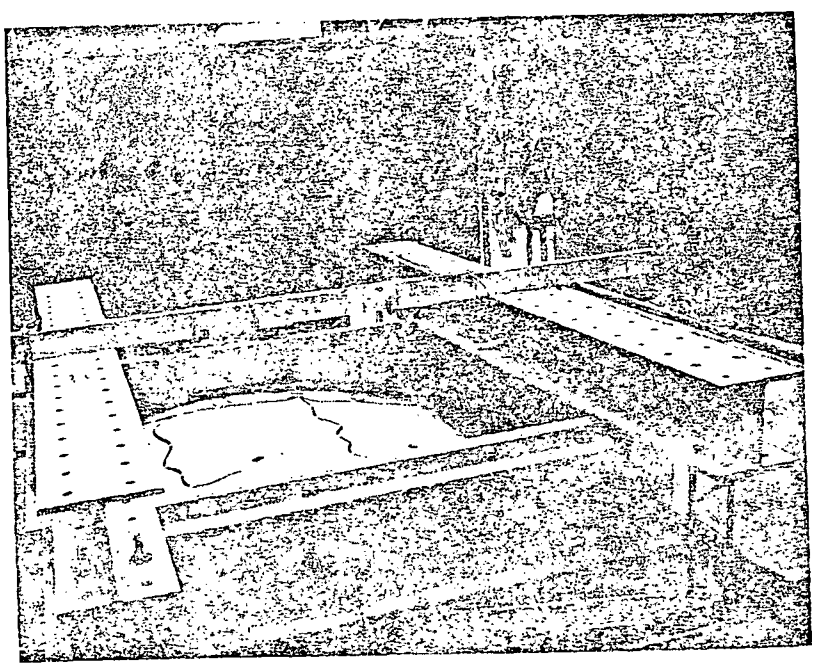


Figure 3-2 Wing-to-Body Fairing Panel in Fixture for Internal Pressurization Test

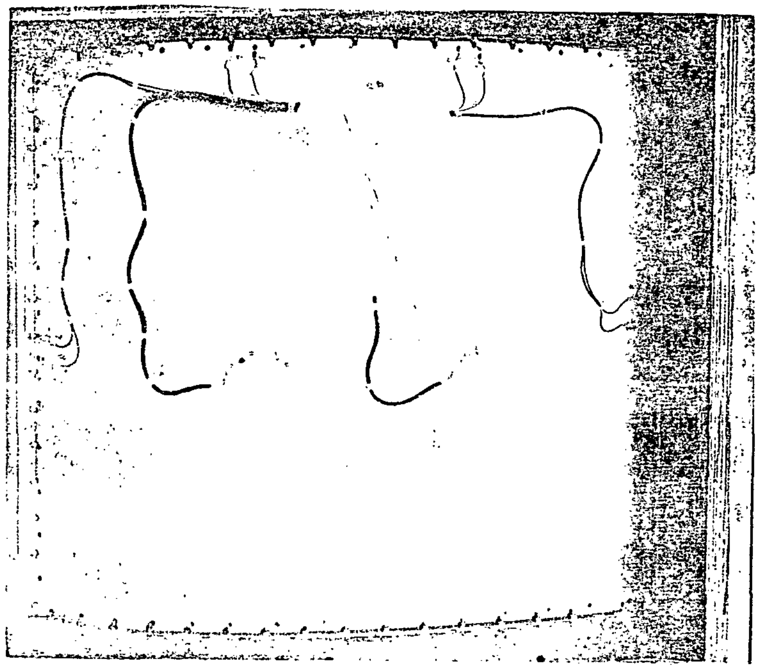


Figure 3-3 Wing-to-Body Fairing Panel in Fixture Showing Compression Skin Failure After External Pressurization Test.

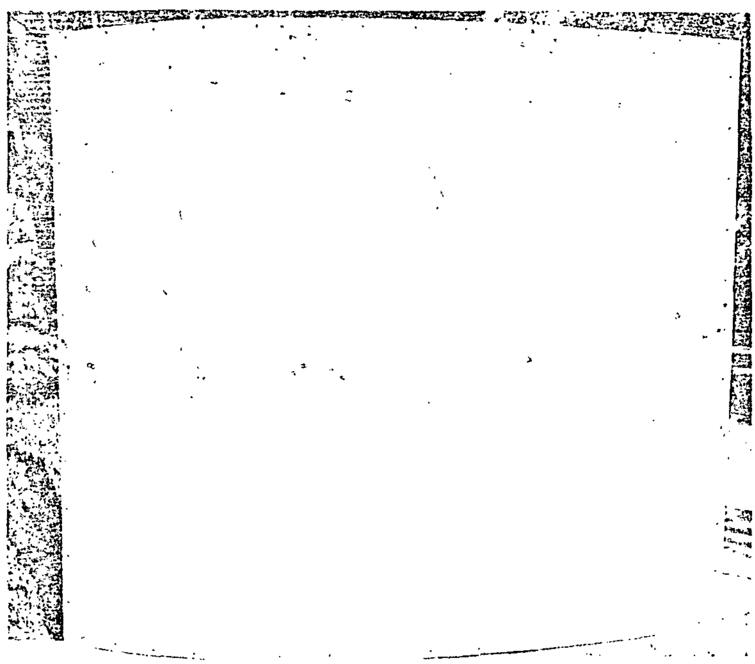


Figure 3-4 Panel Removed From Fixture Showing Compression Skin Failure After External Pressurization Test.

106.18×10^6 newtons/meter^2 (15,400 psi). For the .38 mm (.015 inch) thick PRD-49/epoxy laminate, the preliminary allowable was 104.5×10^6 N/m^2 (15,100 psi). As a result of this failure it was deemed necessary to make all of the flight test PRD-49 wing-to-body fairing panels per the original construction of two plies of the lightweight and one ply of the heavier weight fabric. This was a ply for ply substitution for fiberglass. This construction would provide a margin of safety of +0.14 in compression with a calculated fiber stress of 97.9×10^6 newtons/meter^2 (14,200 psi).

The structural analysis of the three basic parts, the honeycomb wing-to-body fairing panel, the solid laminate wing-to-body fairing fillet and the honeycomb center engine support fairing are given in Appendix A. These analyses have been approved by FAA Designated Engineering Representatives.

4.0 PART FABRICATION

Eight honeycomb wing-to-body fairing panels, (two of these panels were for test), six wing-to-body solid laminate fillets, and six honeycomb sandwich center engine support fairing panels were fabricated at Heath Tecna Corp., Kent, Washington. They also fabricate the basic fiberglass fairings used on the L-1011. Both the wing-to-body honeycomb panels and fillets used PRD-49 impregnated with Hexcel's F-155, 394°K (250°F) cure, 344°K (160°F) service epoxy, whereas, the center engine fairing panels used Hexcel's F-161, 450°K (350°F) cure, 422°K (300°F) service epoxy. All honeycomb was 3.175 mm (0.125 inch) cell size, 48.1 kg/m^3 (3 lb/ft^3) nominal density Nomex® core. In the case of the center engine fairing panels, a layer of compatible adhesive was placed between the core and prepreg to insure structural integrity of the parts. The F-155 system had adequate filleting characteristics to provide a good structural bond between the core and fairing material for the 344°K (160°F) service wing-to-body honeycomb panels.

The process requirements established for each step in the fabrication operation are presented in the following discussion. It should be noted that all of the steps are identical for fiberglass and PRD-49.

When cutting the PRD-49 prepreg, it was found that only about half the number of plies could be cut at once when compared to fiberglass because of the greater toughness of the fibers.

Prior to lay-up, the tools are coated with a water soluble release agent and then a 0.10 mm (0.004 inch) to 0.18 mm (0.007 inch) thick layer of aluminum is sprayed on the tool. The metal spray is then sealed with an appropriate epoxy resin compatible with the ultimate cure temperature of the part. This resin sealer is then gelled at 320°K (115°F) for one hour to facilitate subsequent lay-up. Lay-up of the appropriate number of plies is then done in accordance with the Engineering drawing. Handling of all preimpregnated fabric during cutting and lay-up phases is done under controlled atmospheric conditions with the temperature maintained between 292°K (65°F)

® Du Pont Registered Trademark

and $305^{\circ}K$ ($90^{\circ}F$) and the relative humidity between 20 percent and 65 percent. Following lay-up of the inner skin of honeycomb panels, a ply of .025 mm (.001 inch) Tedlar(R) is applied to the surface of this skin to serve as a moisture barrier. During the lay-up operations, it was determined that one additional step may be required during the manufacture of parts using PRD-49/ epoxy prepreg. With fiberglass prepreg which is somewhat transparent, it is possible to mark a doubler or filler ply as a means of locating the next ply to be laid down. The PRD-49 is not sufficiently clear to permit marking during lay-up, which necessitates a physical measurement to locate each doubler. It would appear, however, that a simple template could be designed which would locate each ply in relation to the core bevel, or the trim line of the part.

No deviation from the specification for bagging and curing was required to produce panels in which PRD-49 was substituted for fiberglass.

Cure of the $344^{\circ}K$ ($160^{\circ}F$) service epoxy system is accomplished in an autoclave using a pressure of $241 \times 10^{3} \pm 3.45 \times 10^{3}$ N/m^{2} (35 ± 5 psi). The temperature is raised from ambient at a rate of $.6^{\circ}K$ ($1^{\circ}F$) to $3.4^{\circ}K$ ($6^{\circ}F$) per minute and then held at $394^{\circ}K$ ($250^{\circ}F$) to $408^{\circ}K$ ($275^{\circ}F$) for one hour minimum. The part is then cooled down to $344^{\circ}K$ ($160^{\circ}F$) maximum under pressure at which time it can be removed from the autoclave. For the $422^{\circ}K$ ($300^{\circ}F$) service parts, the same cure pressure and temperature rise rate is used as for parts cured at $394^{\circ}K$ ($250^{\circ}F$). The cure temperature of $450 \pm 11^{\circ}K$ ($350^{\circ}F \pm 20^{\circ}F$) is maintained for 2 hours minimum and the part cooled under pressure until it reaches $344^{\circ}K$ ($160^{\circ}F$) maximum at which time it is removed from the autoclave. Process control coupons are fabricated with each part using the same materials and simultaneously cured with the part. Test results from the process control specimens for the PRD-49 test panels are provided in Table 4-1. No process controls in addition to those used for fabrication of fiberglass parts were deemed necessary.

Inspection of all parts consists of a visual examination which, in the

(R) Du Pont Registered Trademark

TABLE 4-1. PROCESS CONTROL DATA

Part No.	Wet Compression (RT) Spec. Reqmt.	Wet Compression (RT) Actual N/m^2	Wet Compression (RT) Actual psi	Sandwich Flatwise Tensile (RT) Spec Reqmt.	Sandwich Flatwise Tensile (RT) Actual N/m^2	Sandwich Flatwise Tensile (RT) Actual psi*
1515599-109		141×10^6 149×10^6 139×10^6	20,400 21,600 20,200		2.03×10^6 2.14×10^6 2.10×10^6	295 310 305
1515599-110		155×10^6 146×10^6 145×10^6	22,500 21,200 21,000		2.03×10^6 2.00×10^6 2.14×10^6	295 290 310
1545238-109	131×10^6 N/m^2	137×10^6 139×10^6 138×10^6	19,900 20,200 20,000	1.83×10^6 N/m^2	2.14×10^6 2.17×10^6 2.07×10^6	310 315 300
1545238-110	(19,000 psi)	154×10^6 138×10^6 140×10^6	22,400 20,100 20,300	(265 psi)	1.97×10^6 2.00×10^6 2.21×10^6	285 290 320
1538592-129		152×10^6 138×10^6 142×10^6	22,000 20,000 20,600		2.21×10^6 2.24×10^6 2.14×10^6	320 325 310
1544685-117		135×10^6 137×10^6 141×10^6	19,600 19,900 20,500		2.03×10^6 2.19×10^6 2.10×10^6	295 318 305

* Core Failure

case of PRD-49 panels, is less revealing than for fiberglass panels because of its greater opacity. Therefore, greater emphasis must be placed on NDT procedures. It was found that a tapping procedure used for NDT of fiberglass parts worked equally well on previously fabricated PRD-49 panels. Therefore, it was used for checking panels made for this program. This tapping procedure used a standard aluminum tapper which is bullet shaped at one end and is approximately 12.7mm x 37.7mm (1/2 inch x 1-1/2 inch). The tapper is attached to a handle by a heavy wire. The parts are tapped in a 0.15m (6 inch) grid pattern starting at one corner and working to the opposite corner.

No problems were encountered in the fabrication of any of the parts except for the last left hand center engine fairing panel which partly adhered to the tool. Possible repair procedures were considered but it was decided to scrap the part and use it for further evaluation of machining procedures.

Another part was fabricated and no difficulties were encountered. It should be noted that the same situation has been occasionally encountered in the fabrication of $450^{\circ}K$ ($350^{\circ}F$) cure fiberglass parts. There are several possible causes of this, any one of which might present itself on either glass or PRD-49 parts; namely, (a) insufficient thickness of release agent applied to the tool prior to flame spraying; (b) positioning of the flame spray nozzle too close to the tool, resulting in burn-through of the parting agent; (c) an excessive coating of flame spray coupled with one or both of (a) and (b); or (d) improper application of the gel coat over the flame spray.

In any event, this was an isolated case and should not be cause for concern, except to alert those involved with fabrication to exercise additional care in the preparation of tools involving $450^{\circ}K$ ($350^{\circ}F$) cure temperatures.

Trimming, drilling and countersinking operations for the cured panels are discussed in the Machining Development section of this report.

5.0 MACHINING DEVELOPMENT

Previous experience with PRD-49 in both epoxy and phenolic matrices indicated that this material is very difficult to cut and drill. When the standard fiberglass cutters and drills were used on PRD-49 laminates, tool life was drastically reduced and machined surfaces were badly frayed. In an effort to resolve the problem, a portion of this program was devoted to the development of new machining tools and techniques. The results of this effort are described in detail in Appendix B.

5.1 Trimming and Cutting

Fiberglass reinforced laminates normally are cut and trimmed with diamond coated saws and diamond coated router bits. However, with PRD-49/epoxy laminates these tools rapidly loaded up or became coated with resin and fiber particles. The 2.54 mm (0.10 inch) thick edge laminates on the sandwich panels were overheated and the loose fibers smouldered and occasionally burst into flame. In overheated areas, some delamination was also noted. Figure 5-1 illustrates the fraying that was experienced.

As a result of the development effort at Heath Tecna, special multi-tooth carbide tipped saw blades were designed such that the cutting action would draw the edge fibers downward into the laminate. These saw blades produced the cleanest cut (least amount of fabric fraying), however, the cutting edges dulled quite rapidly when compared to the tool life experienced with cutting fiberglass.

Heath Tecna used a two step operation for cutting the PRD-49/epoxy fairing panels. The initial cut was made with the carbide tipped blades described above and a finish cut to dimension was made with a diamond shaped cut carbide router bit which trims an additional 0.75 - 1.0 mm (0.030-0.040 inch) beyond the initial cut. The routing operation removes the majority of the frayed fibers prior to the final finish deburring (sanding) operation. The two step cut and trim operation adds about 75 percent more labor hours to the cutting time required for fiberglass.

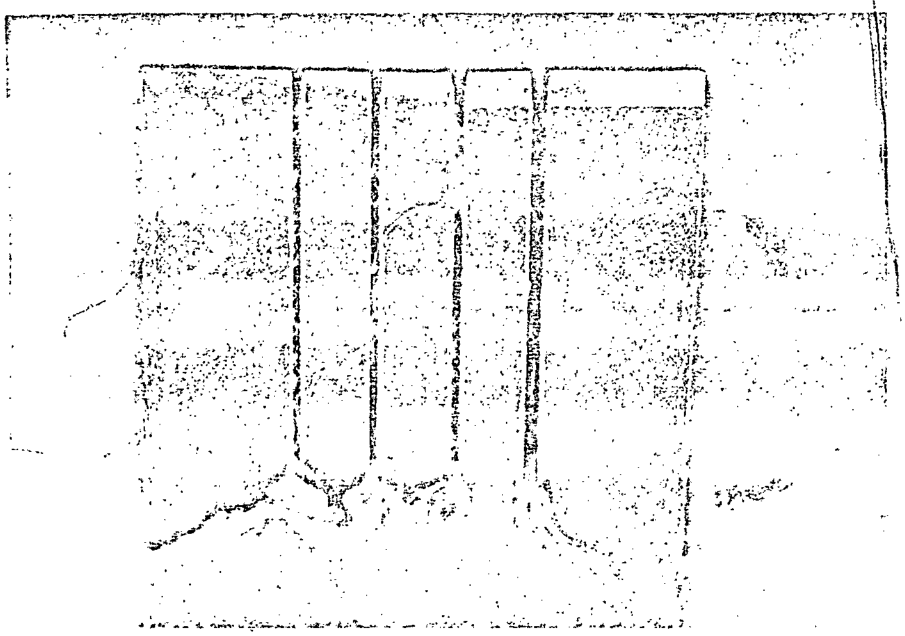


Figure 5-1. Frayed Fibers on PRD-49 Laminate Using Standard Tools for Fiberglass.

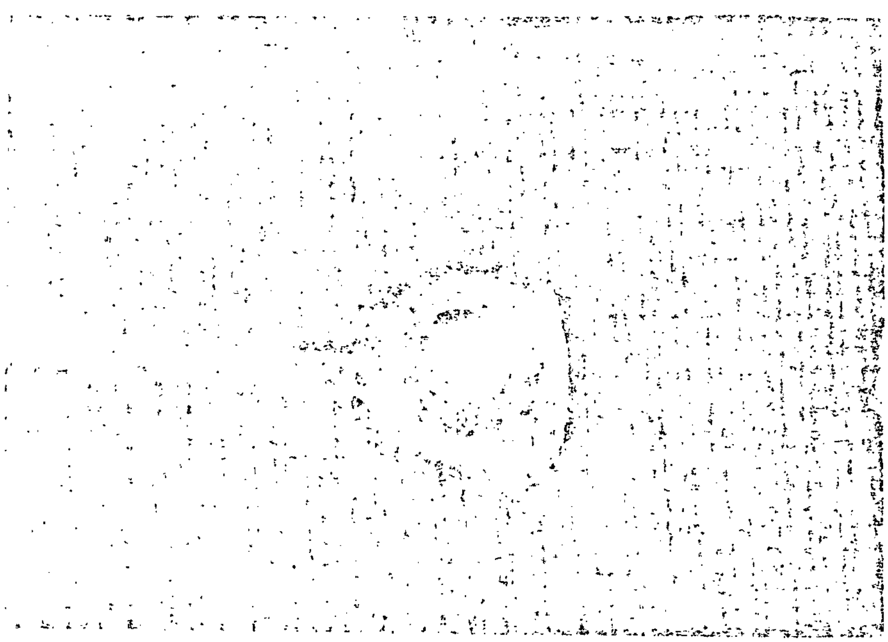


Figure 5-2. Drilled and Countersunk Hole in PRD-49 Laminate Using Standard Tools for Fiberglass.

5.2 Drilling and Countersinking

As with the cutting tools, the standard drills and controlled depth countersink tools presently used for fiberglass laminates were not acceptable for PRD-49/epoxy laminates. These tools produced badly frayed fastener holes and irregular countersinks as shown in Figure 5-2.

Development of an efficient drill point requires a configuration which draws the fibers inward toward the center and cuts them. This approach was also taken in the development of the countersink design. It was also determined that a back-up plate of relatively hard wood or micarta was required to produce a clean hole. Figures 5-3 and 5-4 are samples of holes and countersinks produced by the tools and techniques developed in this program.

The use of back-up blocks will require some additional labor, however, the actual drilling and countersinking operation would be about the same for PRD-49 and fiberglass. Tool life factors could not be determined since long term continuous drilling was not performed.

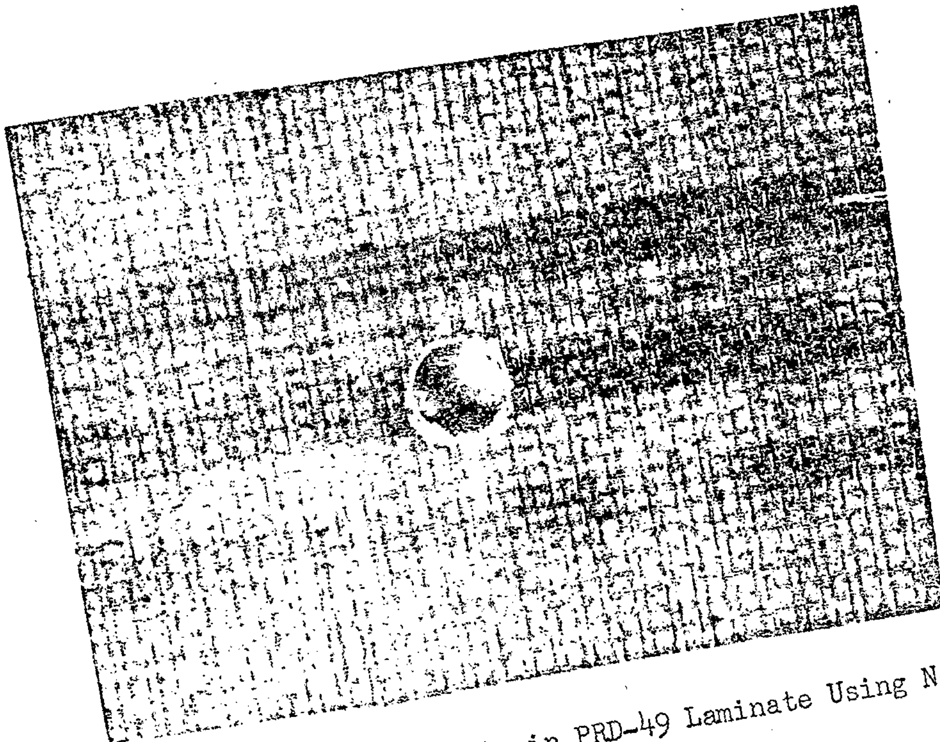


Figure 5-3. Drilled Hole in PRD-49 Laminate Using Newly Developed Drill

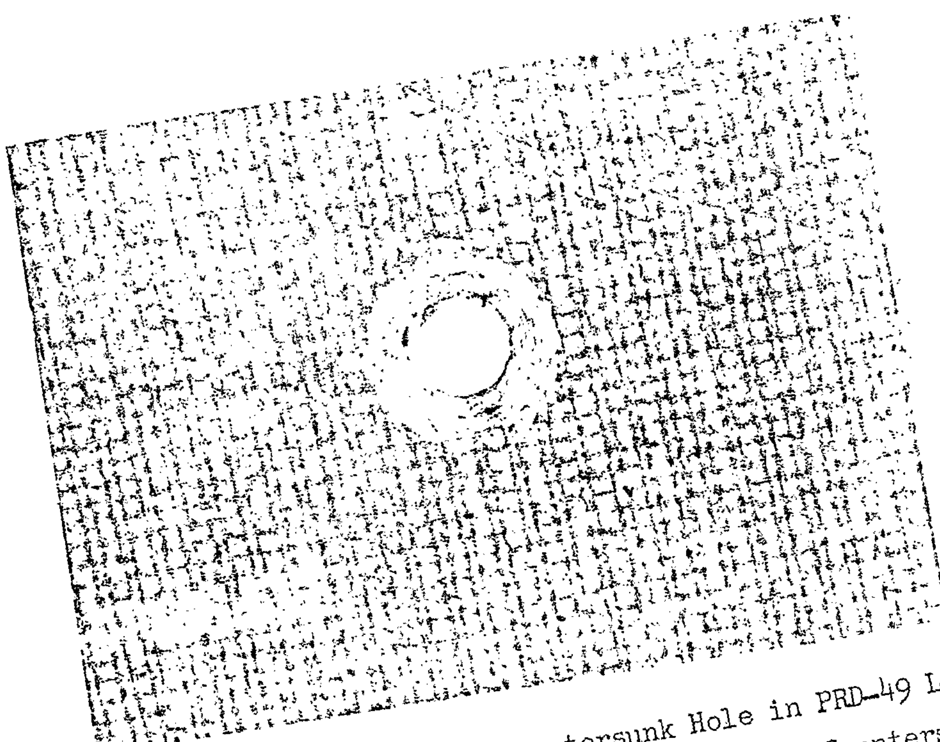


Figure 5-4. Drilled and Countersunk Hole in PRD-49 Laminate Using Newly Developed Drill and Countersink Tool

6.0 PANEL INSTALLATION

During the period of November 6 - 17, 1972, the PRD-49 epoxy fairing panels were installed on two aircraft - Air Canada's airplane, Serial 502, and Eastern Air Line's airplane, Serial 314. The final set of panels were installed on TWA's airplane, Serial 007 in February 1973. Prior to installation of the flight test panels, the regular fiberglass panels were fitted, trimmed, installed on the aircraft, then removed, and deposited in a bonded area. These production fiberglass panels are to be delivered with the aircraft to the airline customers and will later be used as replacements or spares. Eastern Air Line's airplane was delivered January 3, 1973, Air Canada's airplane was delivered February 12, 1973, and TWA's airplane was delivered in March, 1973.

The large wing-to-body fairing sandwich panels were drilled and trimmed net on three sides by Heath Tecna. Each panel was subsequently positioned on the specified aircraft and the hole location and trim line were marked on the unfinished side. (See Figure 6-1). Trimming and drilling of this one side was then completed on the final assembly line. The Porto-Shear described in the Machining Development section (Appendix B) was used for trimming the panels net and minor hand sanding was done to remove the few frayed fibers that remained.

The wing-to-body fillet panels and center engine fairing panels were trimmed net by Heath Tecna but the attach holes were drilled on final assembly after locating on the specified aircraft. A wing-to-body fillet and a center engine fairing panel are shown installed on the aircraft in Figures 6-2 and 6-3, respectively.

Weights of all of the fiberglass and PRD-49 fairing panels were determined after final trimming and drilling. A comparison of the weights, provided in Table 6-1, demonstrates that 7.35 kg (16.2 pounds) per aircraft (26.6 percent) can be saved by using PRD-49/epoxy on these six panels.

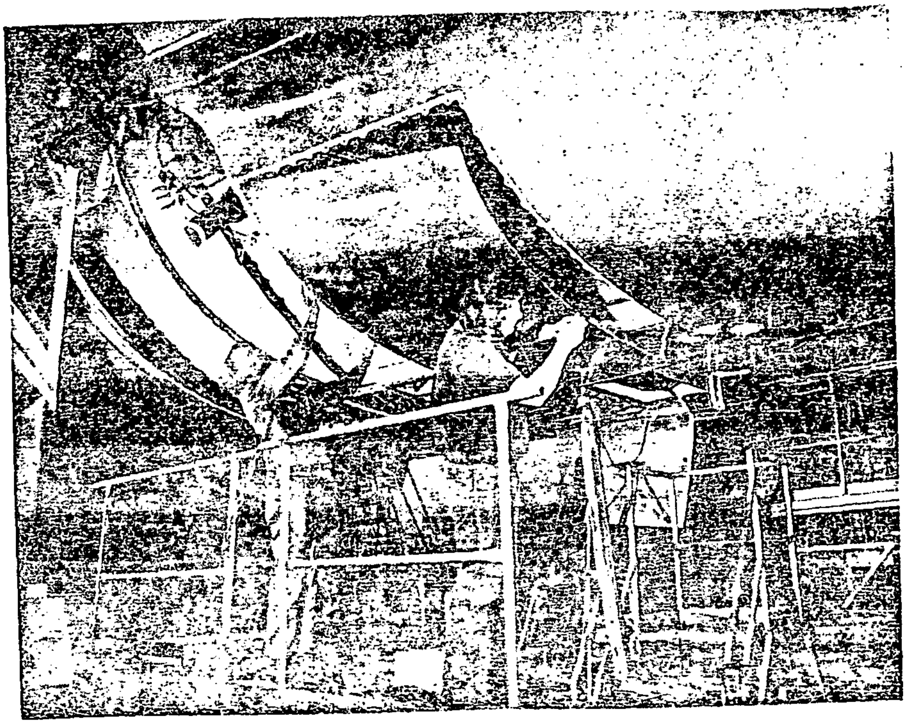


Figure 6-1 Installation of the Wing-to-Body Fairing Panel to Locate Holes and Trim Line Along Bottom Edge

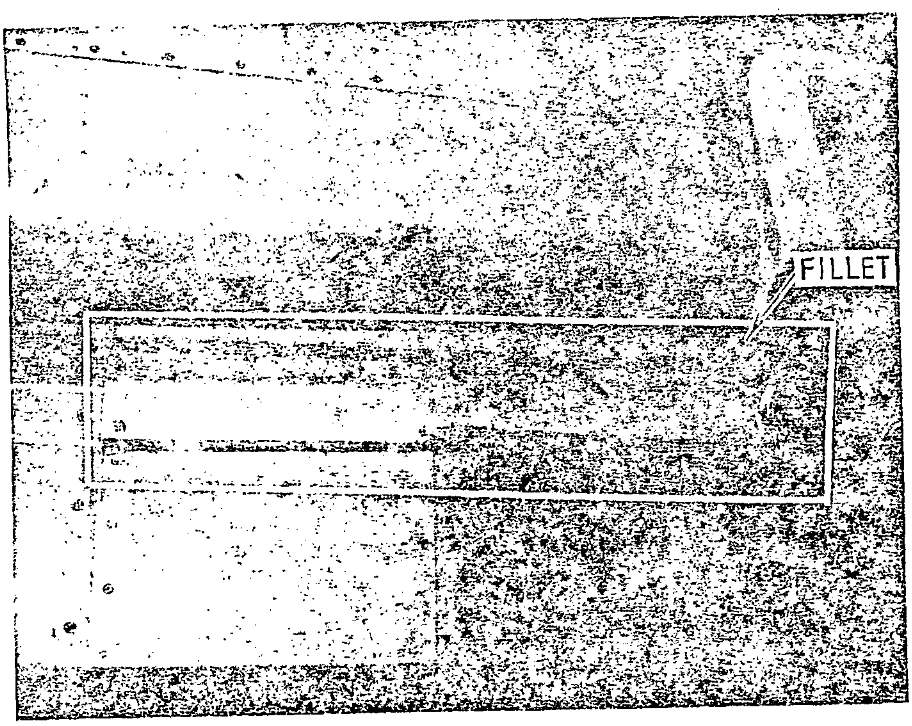


Figure 6-2 Wing-to-Body Fillet Panel Installed on Aircraft

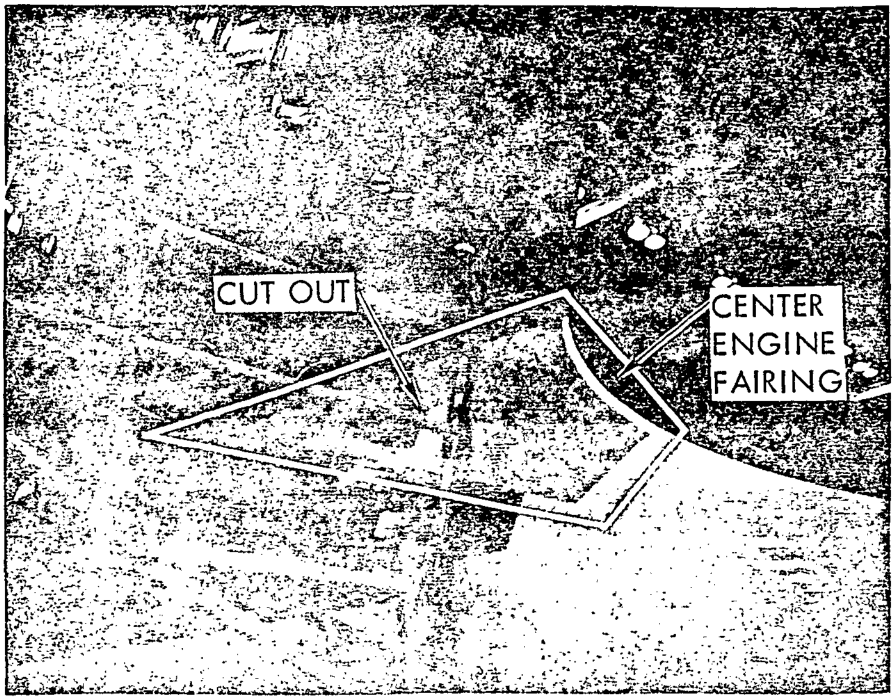


Figure 6-3 Center Engine Fairing Panel Installed on Air Canada Aircraft.

TABLE 6-1. Weight Comparison of PRD-49 and Fiberglass

Part	Fiberglass Weight		PRD Weight		Weight Savings			
					Per Part		Per A/C	
	kg	lbs.	kg	lbs.	kg	lbs.	kg	lbs.
Wing-to-fuselage fairing panel	9.3	20.6	7.0	15.5	2.3	5.1	4.6	10.2
Wing-to-fuselage fillet panel	1.3	2.8	.9	1.9	.4	0.9	.8	1.8
Center engine fairing panel	3.2	7.0	2.2	4.9	1.0	2.1	2.0	4.2
Total	13.8	30.4	10.1	22.3	3.7	8.1	7.4	16.2

Weight Savings - 26.6 percent

7.0 MANUFACTURING COST STUDY

A requirement of the PRD-49 panel fabrication at Heath Tecna was to record the labor hours for each step in the operation. A comparison of the PRD-49 labor hour requirements and the hours for the equivalent fiberglass parts is given in Table 7-1.

Using the figures generated at Heath Tecna, it can be shown that an additional 9.37 hours were required to fabricate one ship set (6 panels) with PRD-49. On a percentage basis this is equivalent to a 15.5 percent increase in labor hours for the PRD-49 panels. Converting these increases into dollars involves assumptions concerning labor rates, burden, scrap rate, general and administrative (G & A) costs and profit. For purposes of this report, the following figures have been used.

Shop Labor	$5.50/hr
Inspection Labor	$6.00/hr
Burden	150 percent of labor costs
Scrap Rate	5 percent
G & A	16 percent
Profit	11 percent

In addition, special tools costing $300 for the three ship sets of parts were required and thus must be added to the overall costs.

The weight saving realized on one ship set of parts was 16.2 pounds or 26.6 percent. Therefore, it can be shown that the above cost elements amounted to an added $37.50 per kilogram ($17.00 per pound) of weight saved.

As discussed in the Machining Development Section (Appendix B), subsequent work at Lockheed demonstrated that trimming time could be reduced with the use of a Black and Decker Porto-Shear. It is estimated that the Porto-Shear could reduce PRD-49 trimming time from the 4.2 hours suggested by Heath Tecna to 3.30 hours per shipset. Since the quality of the cut edge is improved with the Porto-Shear, the deburring time should be about equal to that required for fiberglass. This would reduce the added labor hours from 9.37 for one ship set to 7.55 hours for an increase of 12.5 percent above the fiberglass panels. Using the same labor rates

Table 7-1. Heath Tecna Labor Hour Comparison of Fiberglass vs. PRD-49

Part	No. per Aircraft	Operation														Total all Operations per Part	
		Kit Cutting		Layup		Cure		Trim		Deburr		Assembly (Drill, etc)		Inspection			
		Glass	PRD	Glass	PRD	Glass	PRD	Glass	PRD	Glass	PRD	Glass	PRD	Glass	PRD	Glass	PRD
Wing to Body Fairing Panel	2	.50	.55	7.0	7.6	2.0	2.0	.50	.88	.30	.55	.50	.70	1.0	1.1	11.80	13.38
Wing to Body Fairing Fillet	2	.20	.22	2.0	2.2	2.0	2.0	.35	.61	.12	.20	3.5	4.90	1.0	1.2	9.17	11.33
Center Engine Fairing Panel (LH)	1	.65	.72	4.5	4.9	3.0	3.0	.35	.61	.20	.33	*	*	1.0	1.1	9.70	10.66
Center Engine Fairing Panel (RH)	1	.40	.44	4.0	4.4	3.0	3.0	.35	.61	.20	.33	*	*	1.0	1.1	8.95	9.88

* No holes drilled at Heath Tecna

Table 7-2. Materials Usage

Part	No. per Aircraft	Material per Part - PRD-49			
		0.17 kg/m^2 (5 oz./sq.yd)		0.06 kg/m^2 (1.8 oz./sq.yd)	
		m^2	sq.yd.	m^2	sq.yd.
Wing-to-Body Fairing Panel	2	7.94	9.5	15.0	18
Wing-to-Body Fairing Fillet	2	2.51	3	0	0
Center Engine Fairing Panel (LH)	1	12.12	14.5	5.85	7.0
Center Engine Fairing Panel (RH)	1	12.12	14.5	5.85	7.0

and factors as above, the added costs for fabricating PRD-49 parts is $33.00 per kilogram ($15.00 per pound) of weight saved.

A material cost analysis was made to determine the added costs of using PRD-49. Since there was a direct substitution of PRD-49 for fiberglass, the same amount of prepreg was used for both fabrics. The actual usage for each part is given in Table 7-2. The material costs used in this analysis are shown in Table 7-3. A 10 percent material burden, 5 percent scrap, 16 percent G & A, and 11 percent profit was added to the cost of each fabric.

It was found that the material cost per pound of weight saved is a function of the relative amounts of 0.170 kg/m^2 (5.0 oz./sq.yd.) and 0.061 kg/m^2 (1.8 oz/sq.yd.) materials used. In the case of the wing-to-body fairing panel and wing-to-body fairing fillet the material costs were $113.00 and $123.00 per kg ($51.50 and $56.00 per pound) of weight saved, respectively, whereas the center engine fairing panel calculates to be $303.00 per kg ($137.50 per pound) of weight saved. This points out that the use of PRD-49 in lieu of fiberglass might be best accomplished, at least in the initial phases, on a selective basis where it would be most cost effective for the weight saved. In the case of the center engine fairing panel, the high cost per pound of weight saved may be attributed to the shape of the part and the high usage of the 0.17 kg/m^2 (5.0 oz./yd^2) fabric relative to the usage of the lighter weight fabric. The triangular shape of the part introduces a high trim loss in the prepreg, hence, more material is wasted than for a rectangular part of the same overall dimensions. The high usage of heavy weight fabric may be attributed to the large periphery relative to area of the part. For economy in lay-up time the heavy weight material is used to build up edge thickness. Also, the cut-out area is built up as a solid laminate during lay-up and then the hole cut after cure of the part. See Figure 6-3 for cut-out area.

Table 7-3. PRD-49 and Fiberglass Fabric Costs

Material Thickness Inch	PRD-49		Fiberglass	
	$/m^2	$/sq.yd.	$/m^2	$/sq.yd.
0.254 mm (.010 inch)	$16.15	$13.50	$2.15	$1.88
0.127 mm (.005 inch)	$ 7.18	$ 6.00	$2.79	$2.33

8.0 SPECIAL TEST SPECIMENS

Also included in the program was the fabrication by Heath Tecna of 600 test specimens - 100 short beam interlaminar shear coupons, 100 flexure test coupons, and 100 Celanese type compression specimens using the 0.17 kg/m^2 (5 oz./sq.yd) PRD-49 impregnated with the F-155 resin and a like number using the same fabric impregnated with the F-161 system. These specimens will be tested by the NASA Langley Research Center after a variety of environmental exposures.

9.0 FLIGHT SERVICE EVALUATION

Lockheed has established a program, in cooperation with affected airlines, to gather a five year flight service history on the PRD-49 fairings. After all fairings have been installed on the selected commercial aircraft, annual flight service inspections will be conducted during the routine inspections by the airlines. Reports on the inspections will include sufficient detail to explain comprehensively how the inspections were conducted and results obtained. A schedule of reporting has been established as follows:

Air Canada's aircraft will be inspected yearly in accordance with standard procedures, and a copy of the report will be forwarded to Lockheed.

Eastern Air Lines will keep Lockheed informed as to the location of the aircraft involved at the time of the required inspection. If possible, a member of Lockheed's Maintainability Department will be on location for the inspection. If not, a copy of EAL's inspection report will be submitted through Lockheed's Product Support organization.

The TransWorld Airlines aircraft involved will be inspected at the Los Angeles International Airport by Lockheed Maintainability and Project personnel and the cognizant TWA personnel.

Lockheed will prepare a yearly report on the findings of all three airlines and make the normal report distribution which is stipulated by NASA. A final report on the total program will be issued at the conclusion of the five-year flight service evaluation. It has been determined that PRD-49 panels may be repaired in accordance with the L-1011 Maintenance Manual, Section 51-50-05, 06, and 07, using standard fiberglass repair kits. Accordingly, this information has been entered in the Maintenance Manuals for ACA, EAL, and TWA. Any panels which are damaged beyond repair will be replaced with the spare fiberglass panels. The damaged panels will subsequently be evaluated by Lockheed and NASA.

10.0 RESULTS AND RECOMMENDATIONS

In the initial phase of this program considerable information was obtained in comparing the fabrication characteristics of PRD-49 fabric with fiberglass. Exposing the test panels to commercial airline service for the next five years will indicate the service performance that can be expected from this new fiber.

Prior to receiving FAA approval to fly the PRD-49 panels commercially, material and process control specifications had to be established and the structural analysis had to be confirmed by static tests. The preliminary allowables established for this material and the structural analysis of the panel match the static test results very well. However, more mechanical property data is needed for designing thin skin structure and much more information on environmental effects is required. The six hundred test specimens fabricated in this program for subsequent testing by NASA after environmental exposure and the flight service of eighteen fairing panels will provide much of the needed information.

A portion of this program was devoted to the development of new tools and machining techniques for PRD-49 laminates since the standard fiberglass tools were not acceptable. In cutting the prepreg it was found that only half as many plies of PRD-49 could be cut at one time and the cutting blades dulled much faster. In cutting and drilling the PRD-49 laminates, tools were designed such that the cutting action would draw the edge fibers toward the center of the work. The selected tools successfully cut and drilled the material, however, in many cases the tool life was drastically reduced when compared to the performance of fiberglass tools.

Late in the development program, Lockheed was successful in producing trimming tools for PRD-49 laminates which were almost as efficient as those used on fiberglass. The tool life was increased but the extent could not be measured due to limited usage. More work is required to identify more specifically the life and efficiency of the tools developed in this program.

The lay-up and curing of the PRD-49 parts was no different than the fiberglass since the same resin systems were used. The only problem with PRD-49 epoxy is that it is more opaque than fiberglass, hence it was difficult to locate successive layers during lay-up and about 10 percent more inspection time was required.

After final trimming and drilling in final assembly, the weights of each of the PRD-49 and fiberglass panels were measured. A ship set of six PRD-49 panels resulted in a 7.4 kg (16.2 pound) weight savings or a 26.6 percent reduction of the fiberglass weight.

A manufacturing cost study was conducted on each step in the fabrication process to analyse the cost differential between the labor hours required to produce parts from the two fabrics. After employing the new tooling developments from the program, 7.55 additional hours were needed to produce a ship set of PRD-49 test panels. This amounts to a 12.5 percent labor increase or \$33.00 per kg (\$15.00 per pound) of weight saved. It is believed that time can be reduced even further with experience and additional tool development.

The material cost study showed quite a difference in the added costs for PRD-49 in the various panels. PRD-49 material costs added \$113.00 per kg (\$51.50 per pound) of weight saved in the wing-to-body fairing panel, \$123.00 per kg (\$56.00 per pound) in the wing-to-body fillet, and \$303.00 per kg (\$137.50 per pound) in the center engine fairing panel. This can be attributed to the various mixes of .254 mm (.010 inch) and 0.127 mm (0.005 inch) thick fabrics in the three parts and the excessive amount of prepreg trim that is lost on the center engine panel. The cost differential between the 0.254 mm (0.010 inch) thick fiberglass and PRD-49 is \$14.00 per m^2 (\$11.62 per sq. yd.) whereas the differential between 0.127 mm (0.005 inch) thick material is \$4.39 per m^2 (\$3.67 per sq. yd.) Therefore, the parts with the highest quantity of 0.127 mm (0.005 inch) thick fabric had the lower cost increases.

APPENDIX A

STRESS ANALYSIS DATA FOR FAA APPROVAL OF PRD-49 PANELS

LIST OF SYMBOLS AND ABBREVIATIONS*

BL = buttock line (horizontal distance from vertical centerline plane of aircraft)

C = constant

E = modulus of elasticity

F.S. = fuselage station(horizontal distance from vertical distance plane perpendicular to aircraft centerline)

G_c = modulus of rigidity (shear modulus) of core

I = moment of inertia

M = bending moment

MS = margin of safety

V = dimensionless parameter incorporating sandwich bending and shear rigidities

$V = \frac{\pi^2 D}{b^2 U}$ where D = bending stiffness

U = transverse shear stiffness

(See analyses for particular forms of this equation.)

W.L. = water line (vertical distance from horizontal distance plane of aircraft)

a = panel length

b = panel or beam width

d = sandwich thickness = $t_1 + t_2 + t_c$

h = overall thickness of beam

L = span of beam

p = pressure

t = thickness

y_c = average distance from neutral axis to exterior surface

---

*Special symbols used in this Appendix. Other symbols defined on page vii

σ	=	stress
μ	=	Poisson's ratio
λ	=	$1 - \mu^2$

Subscripts

1	=	outer facing
2	=	inner facing
a	=	property in direction of a
b	=	property in direction of b
c	=	compression when used with E, σ and $\overline{y}$
c	=	core when used with G and t
f	=	flexure
t	=	tension
u	=	ultimate (failing) value

A-1. Wing-to-Body Fairing Panel Analysis

The wing-to-body fairing panel is a honeycomb sandwich. PRD-49 fabric was substituted on a ply for ply basis for fiberglass **in** the panel skins with core thickness and density also the same. The panel is approximately 1.52 x 1.70 meters (60 x 67 inches) and contoured to airplane loft lines. It is a secondary fairing provided to maintain an aerodynamic shape for the lower fuselage forward of the main landing gear compartment. It transfers air loads to a peripheral structure. A sketch of the panel is shown below.

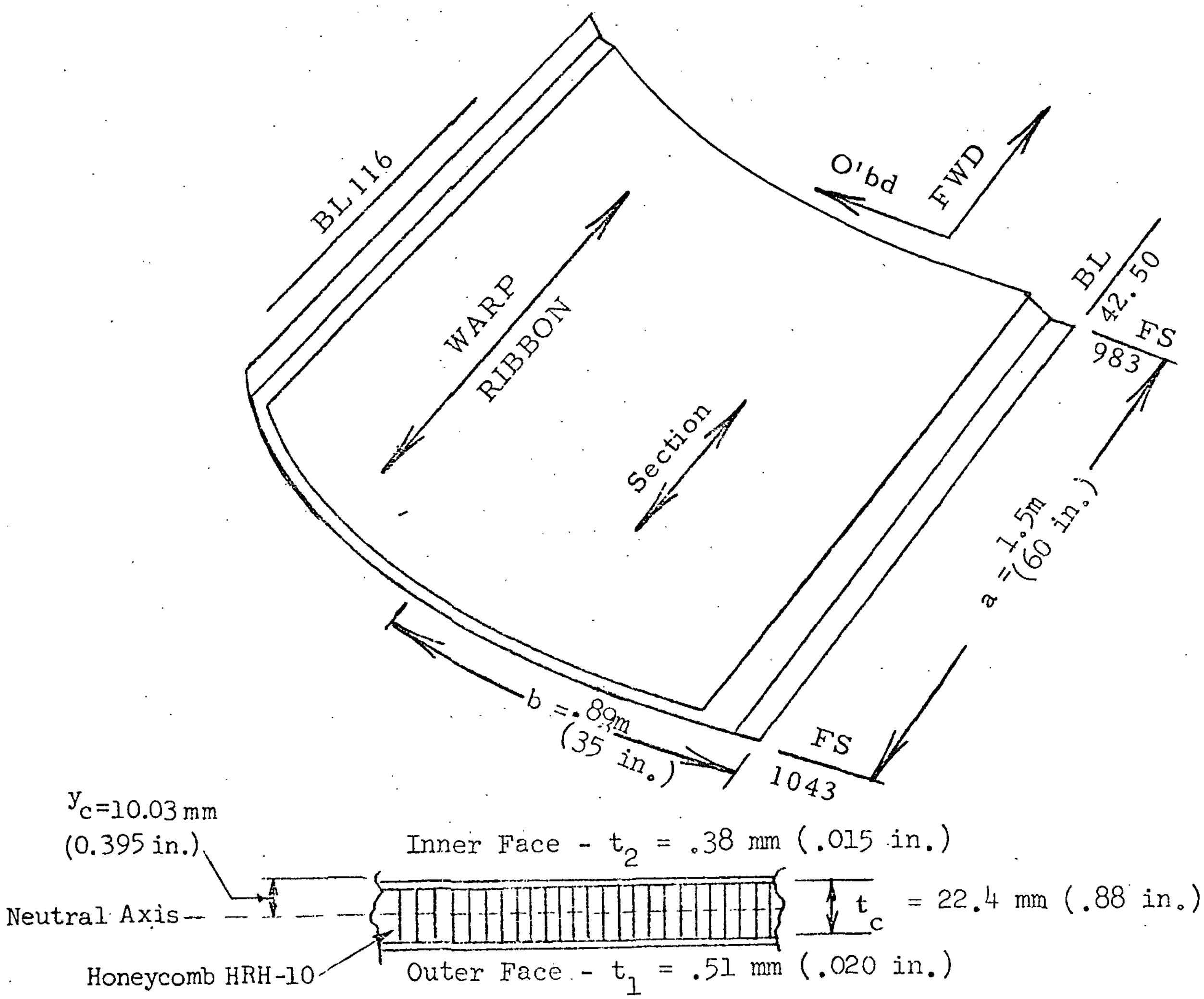


Figure A-1. Wing-to-Body Panel Illustration

This analysis is provided to substantiate the strength of the wing-to-body fairing panels with PRD-49 facings. Buttock line (B.L.) traces through this panel are approximately straight but fuselage station (F.S.) traces show a curvature varying from about 3.81 m (150 in.) radius near buttock line 42.5 to about 1.27 m (50 in.) near buttock line 116. (See Figure A-1.) The critical portion of the panel is the inboard, nearly flat section which is analyzed conservatively as a flat panel using the method outlined in Section 6, Reference 1. The internal loads are 8.27×10^3 N/m^2 (1.2 psi) whereas the external loads are 16.55×10^3 N/m^2 (2.4 psi) so only external loads are considered. This flat portion of the panel is 1.52 m (60 inches) in the "a" direction and 0.889 m (35 inches) in the "b" direction.

Allowables used for PRD-49 laminates are as follows:

$$\sigma_{tu} = 386.9 \times 10^6 \text{ N/m}^2 \ (56.11 \times 10^3 \text{ psi})$$
$$E_t = 25.9 \times 10^9 \text{ N/m}^2 \ (3.76 \times 10^6 \text{ psi})$$
$$\sigma_{cu} = 138.8 \times 10^6 \text{ N/m}^2 \ (20.13 \times 10^3 \text{ psi})$$
$$E_c = 24.9 \times 10^9 \text{ N/m}^2 \ (3.61 \times 10^6 \text{ psi})$$
$$\mu = 0.2$$

The Nomex honeycomb core used in these panels was 3.18×10^{-3} meter (1/8 inch) cell size and 48.1 kg/m^3 (3.0 lb/ft^3) density, and $G_c = 37.9 \times 10^6$ N/m^2 (5500 psi) in the longitudinal direction.

Panel parameters are as follows: (See Figure A-1)

$$t_1 = 0.508 \times 10^{-3} \text{ meter (0.020 inch)}$$
$$t_2 = 0.381 \times 10^{-3} \text{ meter (0.015 inch)}$$
$$t_c = 22.4 \times 10^{-3} \text{ meter (0.88 inch)}$$
$$a = 1.52 \text{ meters (60 inches)}$$
$$b = 0.89 \text{ meters (35 inches)}$$
$$\lambda = (1 - \mu^2)$$

$$I = 2.87 \times 10^{-9} \text{ meter}^4/\text{meter of width } (6.90 \times 10^{-3} \text{ in.}^4/\text{in. width})$$

$$y_c = 10.03 \times 10^{-3} \text{ meter } (0.395 \text{ inch})$$

$$p = 406 \text{ N/m}^2/\text{m width } (2.4 \text{ psi/in. width}) \text{ or } 16.55 \times 10^3 \text{ N/m}^2$$

For sandwiches having unequal faces, the parameter

$$V = \frac{\pi^2 t_c E_1 t_1 E_2 t_2}{\lambda b^2 G_c (E_1 t_1 + E_2 t_2)} \quad \text{Equation 9:2, MIL-HDBK-23A (Ref. 2)}$$

$$= \frac{\pi^2 \times 22.4 \times 10^{-3} \times 24.9 \times 10^9 \times 0.508 \times 10^{-3} \times 25.9 \times 10^9 \times 0.381 \times 10^{-3}}{(1-0.2^2) \times 0.889^2 \times 37.9 \times 10^6 (24.9 \times 10^9 \times 0.508 \times 10^{-3} + 25.9 \times 10^9 \times 0.381 \times 10^{-3})}$$

$$= .0426$$

Using the ratio b/a = .582 and the above value of V = .0426, the constant $C_2 = 0.48$ is obtained from Chart VI-5 of Hexcel Technical Service Bulletin (TSB) 123, (Reference 1) and $C_3 = 0.16$ is obtained from Chart VI-7 of the same reference.

Using these constants the bending moment is calculated across the length using the equations

$$M = \frac{16 pb^2}{\pi^4} (C_2 + \mu C_3) \quad \text{Equation 3a, Section VI of Ref. 1}$$

$$= \frac{16 \times 406 \times 0.889^2}{\pi^4} (0.48 + 0.2 \times 0.16)$$

$$= 27.9 \text{ N/m}^2/\text{m width/meter width } (248 \text{ in.-lb/in. width})$$

$$\sigma_c = \frac{M \bar{y}_c}{I} \quad \text{stress on outer face}$$

$$= \frac{27.9 \times 10.03 \times 10^{-3}}{2.87 \times 10^{-9}} = 97.9 \times 10^6 \text{ N/m}^2 \ (14{,}200 \text{ psi})$$

For laminates 0.508×10^{-3} m (.020 in.) thick, a multiplying factor of 0.8 is used to obtain the allowable facing stress. This factor is based on fiberglass data and is used to calculate the compression facing allowable since the low value of σ_{cu} for PRD-49 makes the panel compression critical.

$$MS = \frac{\sigma_{cu} \times 0.8}{\sigma_c} - 1$$

$$= \frac{138.8 \times 10^6 \times 0.8}{97.9 \times 10^6} - 1$$

$$= 1.14 - 1$$

$$= .14$$

The honeycomb core and the reinforced panel edges are not considered critical in shear.

A-2. Wing to Body Fairing Fillet Analysis

This part is a solid laminate and PRD-49 was substituted on a ply for ply basis for fiberglass in its construction. The panel has a configuration as shown in the sketch below.

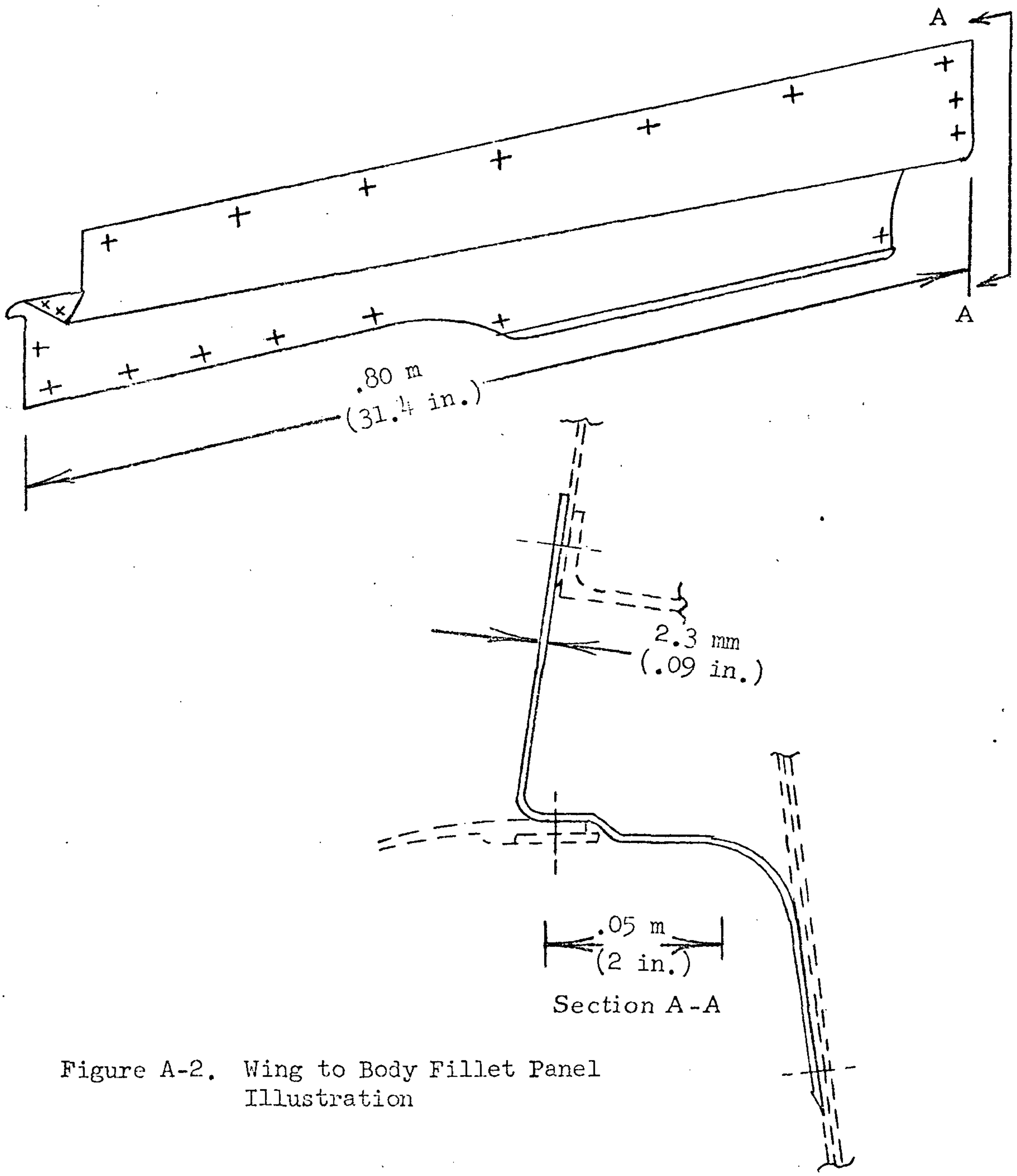


Figure A-2. Wing to Body Fillet Panel Illustration

Because of the angular shape of this fillet, it is not considered critical as a beam between the end supports. The longest free diagram length is less than 0.102 meter (4 inches) but for purposes of calculation this dimension is used. The analysis is based on the assumption that the part is loaded as a simply supported rectangular beam with uniform loading.

Since PRD-49 is compression critical, the margin of safety is calculated only for this mode using

$$\sigma_{cu} = 138.8 \times 10^6 \ N/m^2 \ (20.13 \times 10^3 \ psi)$$

Panel parameters for the analysis are as follows:

$p = 350 \ N/m^2/m \ (2.0 \ psi/in.)$

$L = 0.102$ meter (4 inches)

$h = 2.286 \times 10^{-3}$ meter (0.09 inch)

$b = 2.54 \times 10^{-2}$ meter (1.0 inch) since analysis is done per inch of width

$y_c = h/2 = 1.143 \times 10^{-3}$ meter (0.045 inch)

Using these values the maximum bending moment is calculated as

$$M_{max} = \frac{pL^2}{8}$$

$$= \frac{350 \times 0.102^2}{8}$$

$$= 0.452 \ \text{N-m/m width (4 inch-pounds/in.)}$$

The moment of inertia of a rectangular beam is given by

$$I = \frac{bh^3}{12}$$

$$= \frac{2.54 \times 10^{-2} \times (2.286 \times 10^{-3})^3}{12}$$

$$= 25.3 \times 10^{-12} \ \text{meter}^4 \ (61 \times 10^{-6} \ \text{inch}^4)$$

The maximum fiber stress is calculated using the equation

$$\sigma = \frac{My_c}{I}$$

$$= \frac{0.452 \times 1.143 \times 10^{-3}}{25.3 \times 10^{-12}}$$

$$= 20.4 \times 10^{6} \text{ N/m}^2 \text{ (2960 psi)}$$

The thickness correction factor for a 2.286×10^{-3} meter (0.09 inch) laminate based on fiberglass data is 0.95.

$$MS = \frac{\sigma_{cu} \times .95}{\sigma_c} - 1$$

$$= \frac{138.8 \times 10^{6} \times .95}{20.4 \times 10^{6}} - 1$$

$$= 6.46 - 1$$

$$= 5.46$$

A-3. Center Engine Fairing Panel

The center engine fairing panel is a honeycomb sandwich. PRD-49 was substituted on a ply for ply basis for fiberglass on the panel skins with the core and panel dimensions remaining constant. The panel is roughly triangular in shape and contoured to aircraft loft lines. Its maximum dimensions being approximately 0.76 meters by 1.83 meters (30 x 73 inches). It is located above the center engine and maintains an aerodynamic shape for the support structure for this engine, transferring air loads to this structure. A sketch of the panel is shown below.

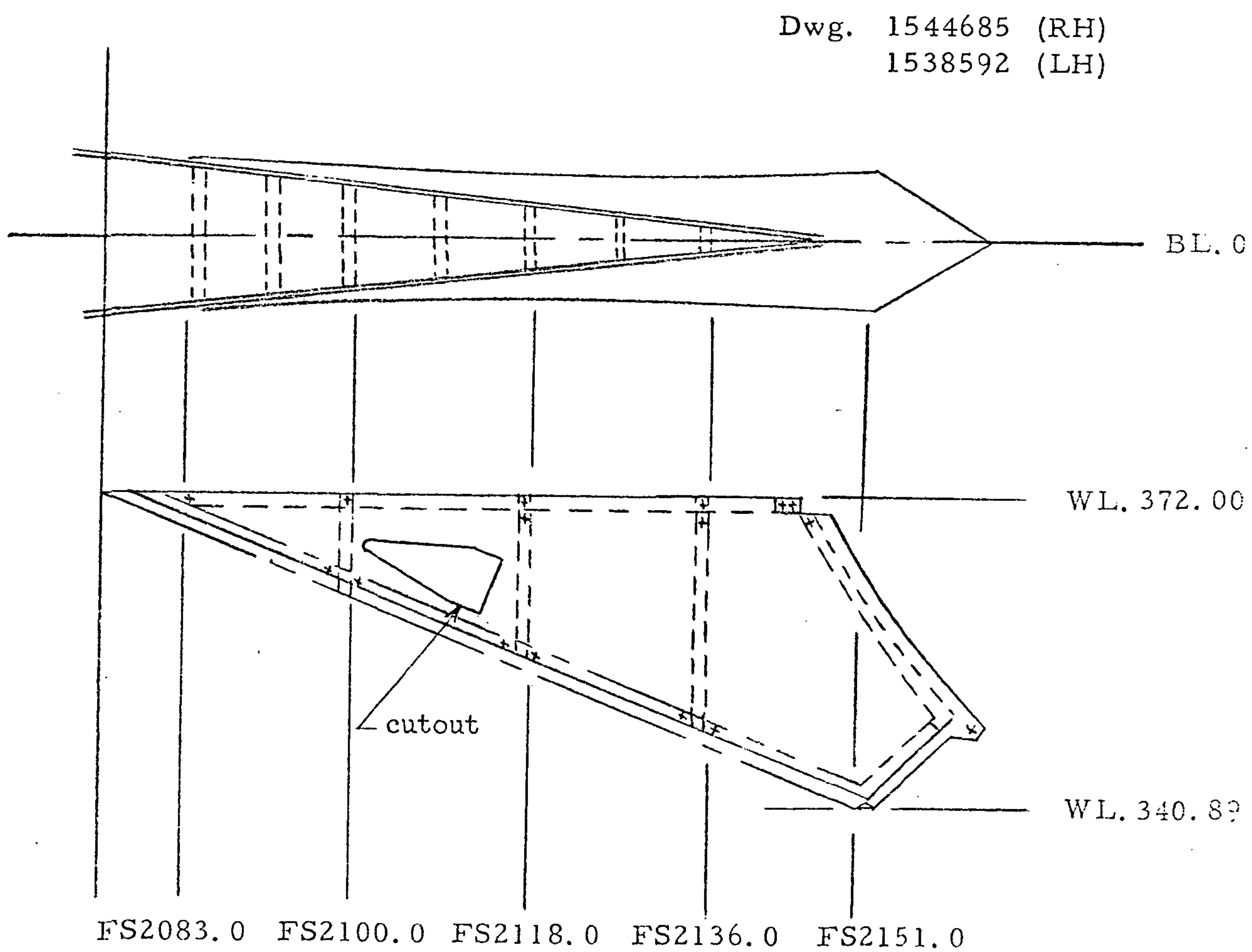


Figure A-3. Center Engine Fairing Panel Illustration

The panel is attached to the structure around its periphery and also to intermediate frames at the fuselage stations indicated in Figure A-3. Because of the attachment to the intermediate frames, it can be considered as a series of smaller panels. For this analysis one such panel extending between fuselage stations 2118.0 and 2136.0, with its top at water line 372.0, is considered. For simplicity, the panel is considered as being rectangular in shape, simply supported along all four edges as shown in Figure A-4.

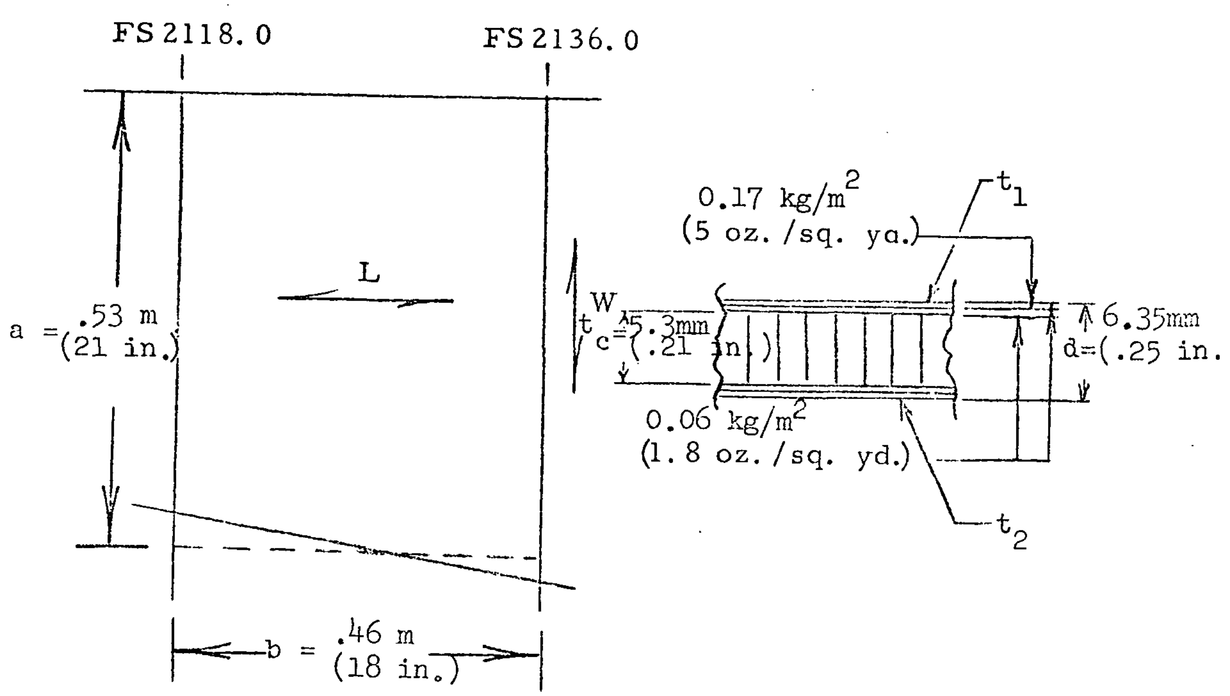


Figure A-4. Center Engine Fairing Panel Section

The internal and external design air loads are the same and equal to 10.34×10^3 N/m^2 (1.5 psi). Since PRD-49 is critical in compression, the analysis is based on the compressive properties of the material as given below:

$$\sigma_{cu} = 138.8 \times 10^6 \text{ N/m}^2 \ (20.13 \times 10^3 \text{ psi})$$

$$E_c = 24.9 \times 10^9 \text{ N/m}^2 \ (3.61 \times 10^6 \text{ psi})$$

$$\mu = 0.2$$

The Nomex honeycomb core used in these panels was 3.18×10^{-3} meter (1/8 inch) cell size, 48.1 kg/m^3 (3.0 lb/ft^3) density and G_c = 15.17×10^6 N/m^2 (2200 psi) in the transverse direction.

Panel parameters are as follows:

$$t_1 = t_2 = t = 0.508 \times 10^{-3} \text{ meter (0.020 inch)}$$

$$t_c = 5.33 \times 10^{-3} \text{ meter (0.21 inch)}$$

$$a = .533 \text{ meter (21 inches)}$$

$$b = .457 \text{ meter (18 inches)}$$

$$d = 6.35 \times 10^{-3} \text{ meter (0.25 inch)}$$

$$p = 10.35 \times 10^3 \text{ N/m}^2\text{/m width (1.5 psi/in. width)}$$

For sandwiches having equal facings, the parameter

$$V = \frac{\pi^2 E t t_c}{2 \lambda b^2 G_c} \qquad \text{Equation 9:2a, MIL-HDBK-23A (Ref. 2)}$$

$$= \frac{\pi^2 \times 24.9 \times 10^9 \times 0.508 \times 10^{-3} \times 5.33 \times 10^{-3}}{2(1 - 0.2^2) \times 0.457^2 \times 15.17 \times 10^6}$$

$$= 0.109$$

Using the ratio b/a = .877 and the above value of V = 0.109, the constant C_2 = 0.25 is obtained from Chart VI-5 of Hexcel TSB 123 (Reference 1) and C_3 = 0.23 is obtained from Chart VI-7 of the same reference.

Using these constants, the bending moment is calculated across both dimensions of the panel using the indicated equations

$$M_a = \frac{16\ pb^2}{\pi^4}(C_3 + u\ C_2)$$ Equation 3, Section VI, Reference 1, for moment across width

$$= \frac{16 \times 10.35 \times 10^3 \times (.457)^2}{\pi^4}(0.23 + 0.2 \times 0.25)$$

$$= 358 \times 0.28$$

$$= 100 \text{ N-m/m } (22.4 \text{ in.-lb/in.})$$

$$M_b = \frac{16pb^2}{\pi^4}(C_2 + \mu C_3)$$ Equation 3a, Section VI, Reference 1, for moment across length

$$= 358\ (0.25 + 0.2 \times 0.23)$$

$$= 358 \times 0.296$$

$$= 109 \text{ N-m/m } (24.3 \text{ in.-lb/in.})$$

Since M_b is greater than M_a, M_b is used to calculate the panel facing stress using the following equation

$$\sigma_c = \frac{2\ M_b}{t(d+t_c)}$$ Equation 4, Section VI, Reference 1 for facing stress in equal thickness sandwich

$$= \frac{2 \times 109}{.508 \times 10^{-3}\ (6.35 \times 10^{-3} + 5.33 \times 10^{-3})}$$

$$= 36.9 \times 10^6 \text{ N/m}^2 \text{ (5340 psi)}$$

The temperature of the structure at take-off is approximately 55°C (130°F). At this temperature it is assured that the laminate will retain 90% of its room temperature strength. The thickness multiplying factor is 0.8 for a 0.508×10^{-3} meter (0.020 inch) laminate.

$$MS = \frac{\sigma_{cu} \times .9 \times .8}{\sigma_c} - 1$$

$$= \frac{138.8 \times 0.9 \times 0.8}{36.9} - 1$$

$$= 2.70 - 1$$

$$= 1.70$$

References

1. Hexcel Technical Service Bulletin 123, Design Handbook for Honeycomb Sandwich Structures, Hexcel Corp., Dublin, Calif., March 1970

2. Military Handbook 23-A, Structural Sandwich Composites, Department of Defense, Washington, D.C., 30 Dec. 1968

APPENDIX B

MACHINING DEVELOPMENT

The following is a summary of the results of the Heath Tecna and Lockheed machining development program.

B-1. Prepreg Cutting Procedures

The initial operation in the fabrication of composite sandwich or solid laminate parts is cutting the prepreg material into various patterns for face sheets, doublers or fillers which are then arranged into kits and stored. Fiberglass prepreg material can be stacked up to 50 plies thick for cutting but it was found that only half as much PRD-49 prepreg material could be cut. Most bond shops utilize a standard Stanley carton knife (Type #1299, Blade #1992) for cutting prepreg materials.

Blade changes are relatively infrequent when cutting fiberglass prepreg but when cutting PRD-49 prepreg close attention must be paid to the condition of the cutting edge as any nicks will tend to catch on the fabric and cause fraying. The added care required and the fact that less plies can be stacked for cutting cause an increase in labor costs of approximately 10 percent.

Recent tests conducted at Lockheed using an X-Acto No. 28 blade demonstrated that 32 plies of PRD-49/epoxy prepreg material could be stacked and cut with this blade. The life of the X-Acto blade was considerably better than that of the Stanley blade.

B-2. Laminate Trimming and Machining

Fiberglass reinforced laminates normally are trimmed and machined with air driven motor saws and routers and electric motor driven routers. To cut and trim the L-1011 fiberglass fairing panels, Heath Tecna used #40 and #80 grit diamond coated 76.2 mm (3 inch) and 101.6 mm (4 inch) diameter saws and 6.35 mm (0.25 inch) to 25.4 mm (1.0 inch) diameter router bits. Minor hand sanding was used to finish the edges.

The standard diamond coated saws and router bits were evaluated in the initial test on PRD-49 epoxy laminates. The saws were driven by air motors at 16,000 rpm and the router bits were driven at 35,000 rpm.

Grit size on the cutting tools had no effect upon the surface produced, as the tools rapidly became "loaded-up" (coated) with resin and fiber particles. The 2.54 mm (0.10 inch) thick edge laminates on the sandwich panels were overheated, the loose fibers smouldered and occasionally burst into flame, and some delamination was also noted.

Various saw blade and router bit configurations were evaluated at Systimatic Tool Company, Seattle, Washington. Laminates made from 394°K (250°F) and 450°K (350°F) curing epoxy resins were used for the machining tests.

Carbide tipped saw blades were designed such that the tooth would draw the edge fibers downward into the laminate as the cutting action occurred. The saw blades depicted in Figures B-1 through B-3 produced the best results on PRD-49 epoxy laminates. The blades depicted in Figures B-1 and B-2 produced the cleanest cut (least amount of fabric fraying), however, the cutting edges dulled rapidly and generated more heat than the saw blade shown in Figure B-3.

Although blade velocity did not have a significant effect upon the finish of the cut edge, the tool life was affected. A speed of 6000 rpm was better than 16,000 rpm in deterring the rather rapid dulling of the carbide cutting tips.

Depending on the frequency of use of these tools, resharpening could be required after each day's production run. This necessitates a greater quantity of tools and a regular, controlled sharpening schedule. With fiberglass laminates a diamond coated saw or router bit lasts 6-8 months when used 4-6 hours per day, five days a week.

The cutting system selected by Heath Tecna as being the most feasible at this time is a two step operation. The initial cut is made with the blade shown in Figure B-3, so that an excess of 0.76-1.02 mm (0.030-0.040 inch) beyond the standard setback is added to the laminate panel edge.

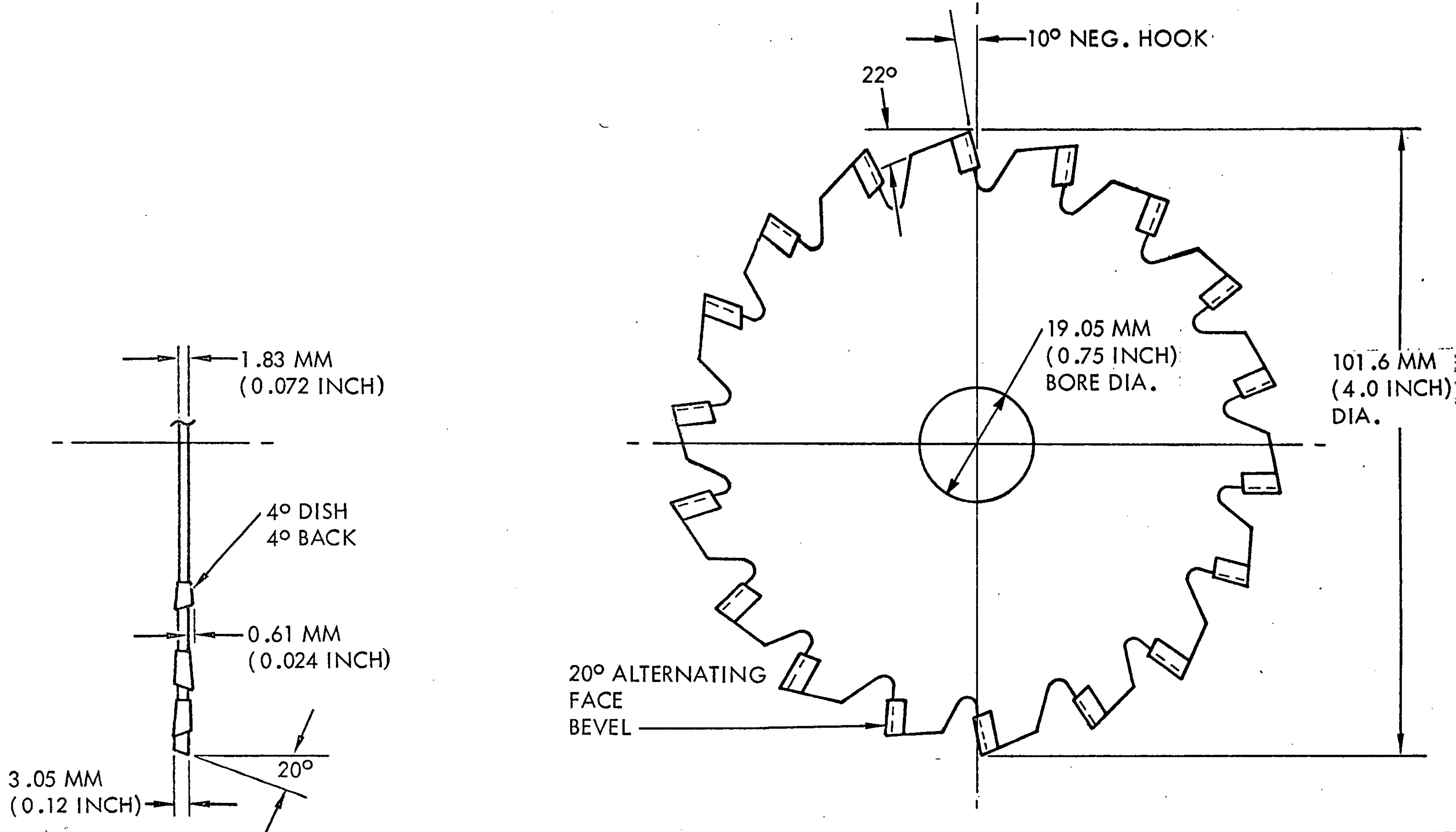


Figure B-1. Carbide Saw 18-Teeth

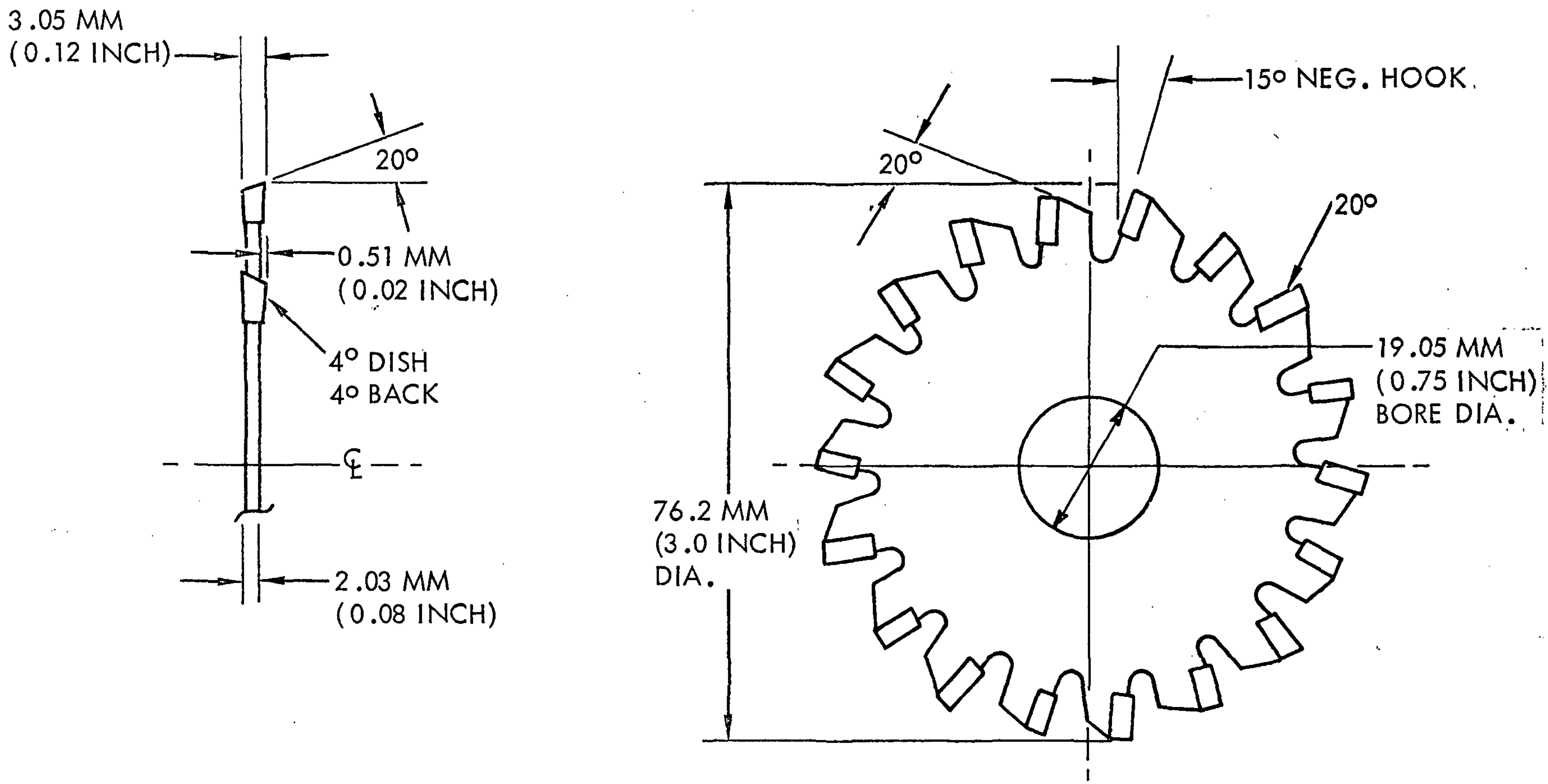


Figure B-2. Carbide Saw 18-Teeth

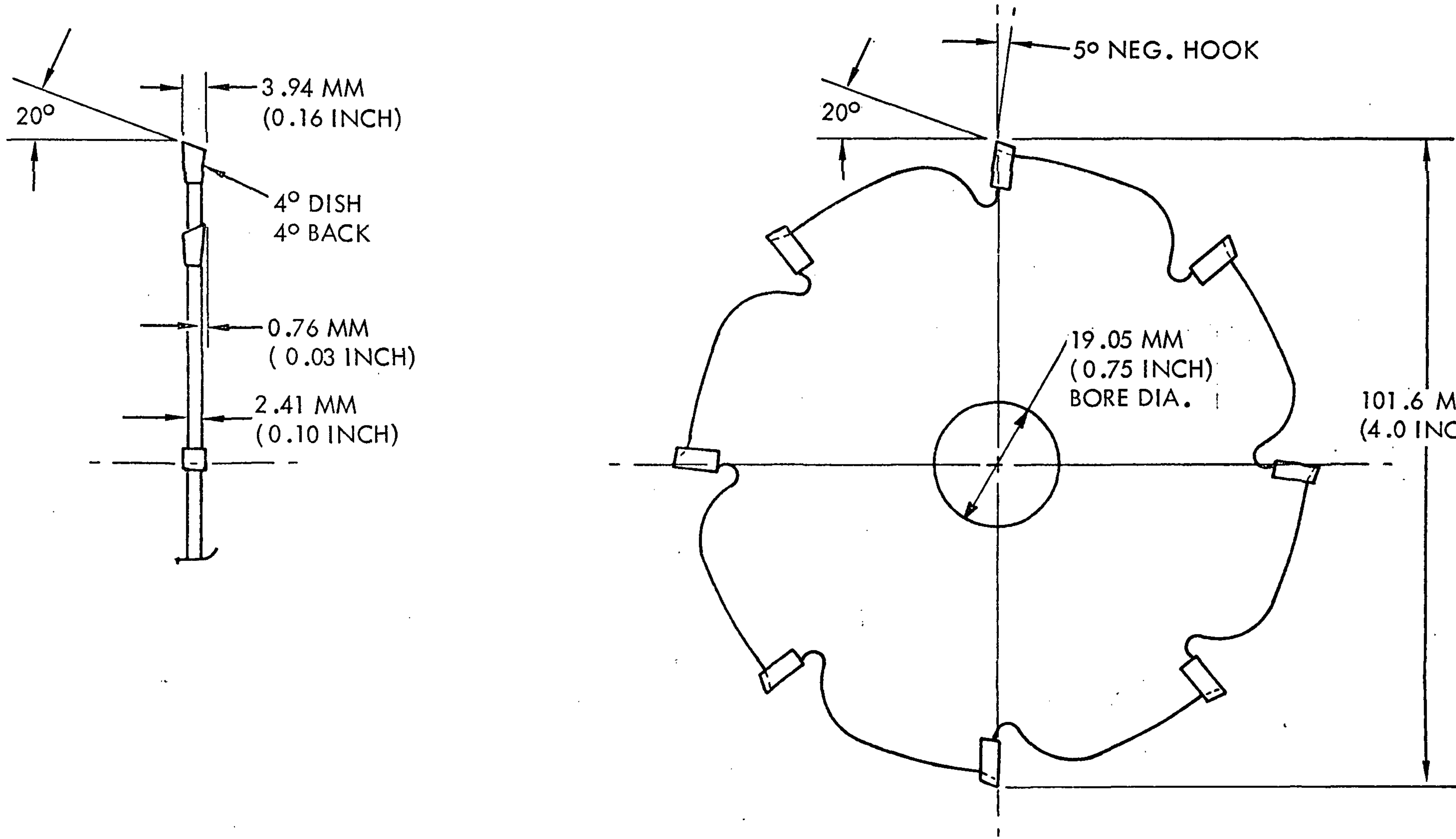


Figure B-3. Carbide Saw 8-Teeth

The final finish cut to dimension is accomplished with a hand air motor and a diamond cut carbide router bit, operating at 35,000 rpm. The routing operation removes the majority of the frayed fibers prior to the final finish deburring (sanding) operation. This two step trim operation adds 75 percent more labor hours to the trimming time, but it reduces the additional labor hours required during final deburring. The increase does not include the cost for additional cutters (approximately $50 each) or the added resharpening costs (about $7.00) however, these costs are a small increment when compared to the labor increase.

Hand routing is also used to cut sharp radius areas and cutouts in the sandwich and solid laminates. When solid carbide end mill cutters were evaluated for routing, they were found to overheat the PRD-49 epoxy laminate and dull rapidly. Best results were obtained with diamond shaped cut carbide cutters.

Liquid or gas coolants were not evaluated during this program because of the difficulty involved in adapting a continuous flow system to a hand tool operation. Furthermore, the introduction of liquid coolants may have a detrimental effect on the laminate due to absorption of the coolant.

The deburring operation is performed after trimming and machining is complete. Fiberglass reinforced laminates require only a quick scuff hand sand for final finishing, however, it appears machine sanding may be necessary after routing PRD-49 epoxy composites. A wet and dry vibrating sander and 100-180 grit sandpaper was used by Heath Tecna. This incurred a twenty percent increase in the final deburring operation.

Previous independent work at the Lockheed-California Company has lead to the development of a proprietary tool identified as a "Nibbler" which incorporates a very close tolerance shearing operation for trimming contoured laminates. A comparison of cuts by the Nibbler and a standard fiberglass router is shown in Figure B-4. Two models, a light duty unit

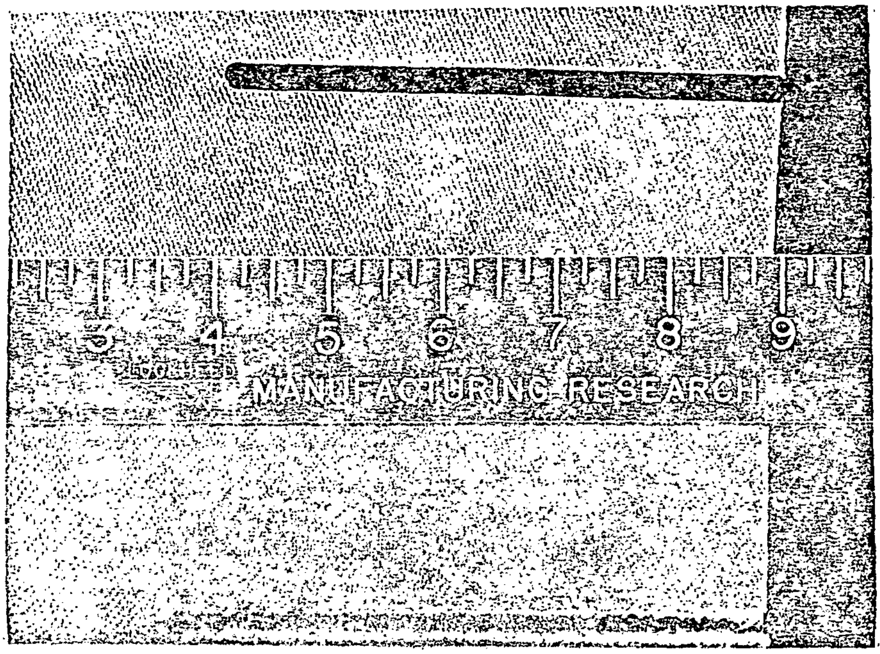


Figure B-4 - Comparison of Cuts by Nibbler (Top) and Standard Fiberglass Router (Bottom). Scale shown is in inches.

for thinner laminates, and a heavy duty unit, have been produced and are available as standard Lockheed tools. Lockheed is taking steps to make this tool available under license.

The heavy duty unit successfully trims laminates up to 3.18 mm (0.125 inch) thick. However, the force required to push the Nibbler in this thick material would fatigue an operator using it all day. It is best suited for cutouts, and sharp radii and is not recommended for long cuts. Both units have proven to be better adapted to epoxy resin systems than to phenolics. This can probably be attributed to the better wetting characteristics of the epoxies, which create a better bond between fibers and resin. Sanding requirements following trimming with the Nibbler are minimal.

A second tool, evaluated during the installation of panels on Air Canada and Eastern Air Lines aircraft at Palmdale, is the Black and Decker Porto-Shear. (See Figures B-5 and B-6). This is a hand held electric motor driven device utilizing two opposing shear blades which operate on the jigsaw principle. These units come in 14, 16, and 18 gage models and are available commercially from Black and Decker. A similar model is available from Rockwell Air Tools. The 16 gage model was used for the trimming operation on the fairing panels. The 60 inch edge of the panels was trimmed in 90 seconds and required little or no sanding after trim. There was insufficient material cut to evaluate the life of the shear blades, however, the cutting and sanding time for PRD-49 and fiberglass laminates was the same with the Porto-Shear.

Based on the above evaluation, Lockheed used the Nibbler and Porto-Shear to trim heavier PRD-49/epoxy laminates on all of the flight evaluation panels. Further evaluation of the two tools will be conducted to determine tool life.

B-3. Drilling and Countersinking

Hole drilling and countersinking operations on the L-1011 fairings were accomplished during the fit check of trimmed panels to the aircraft contour fixtures at Heath Tecna. The standard drills and controlled depth countersink tools presently used for fiberglass laminates were not acceptable for PRD-49/epoxy laminates.

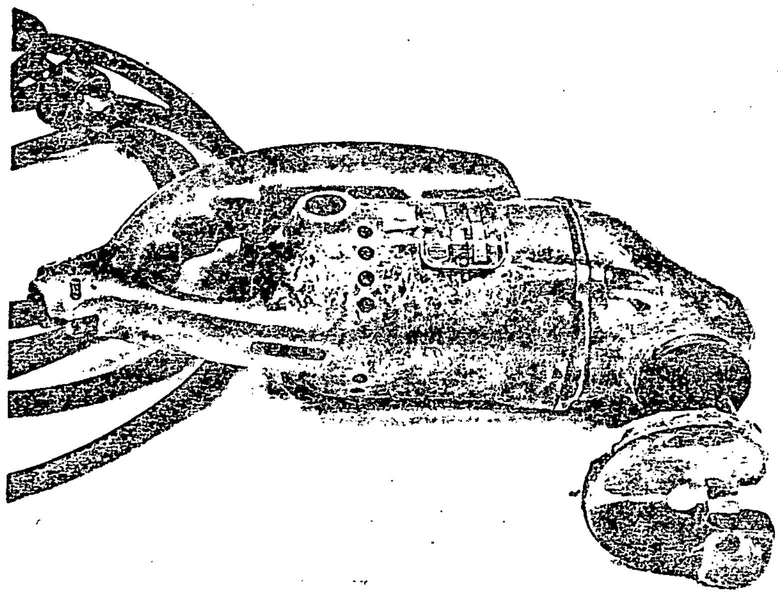


Figure B-5 - Black and Decker Porto-Shear Used
To Trim Wing-to-Body Fairing Panel
During Installation

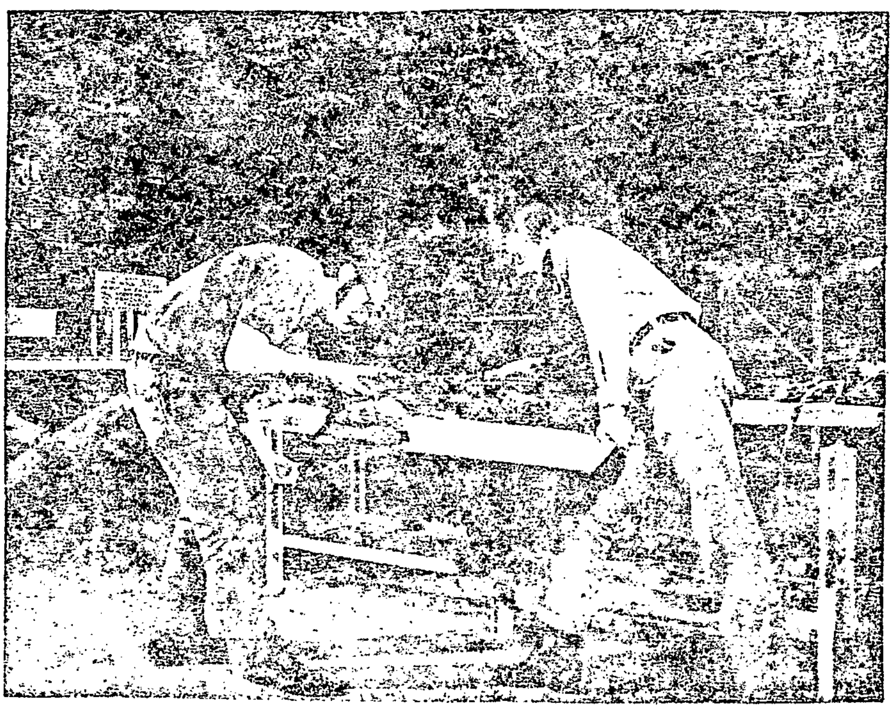


Figure B-6 - Trimming Wing-to-Body Fairing
Panel With Porto-Shear

These tools produced badly frayed fastener holes and irregular countersinks. Drill speed variations from 500 to 5,000 RPM were of no significant effect.

Development of an efficient drill point requires a configuration which draws the fibers inward toward the center and cuts them. This approach was also taken in the development of the countersink design.

The drill point defined in Figure B-7 was selected for drilling holes in PRD-49 laminates. When backed up properly, a clean hole was produced with this type of drill. Backup material may be provided by clamping a strip of laminate, relatively hard wood or micarta on the opposite side of the panel from which the drill enters. All drilling was performed with a standard drill motor operating at 5000 RPM.

While the special drill just described produced acceptable holes, all the holes drilled at Lockheed during installation of the panels were drilled with a Lockheed Standard Stepped Double Margin Drill, shown in Figure B-8. These drills produced clean sharp holes when properly backed up and may be reground until minimum dimensions are reached. Dimensions of the drills are controlled by NAS 937.

The use of backup blocks will require additional labor for positioning and seating the blocks. Tool life factors could not be determined since long term continuous drilling was not performed.

<u>Countersinking</u> - Two countersink configurations were developed during the program. The tool in Figure B-9 is a general configuration and is applicable to single or multiflute tools. This tool was successfully used at Heath Tecna for countersinking the test panels. The standard 100° stop countersink is placed in a stationary chuck and the 20° relief is ground out by hand with a drill motor and diamond stone cutter. In a production tool this cut would be made prior to heat treat.

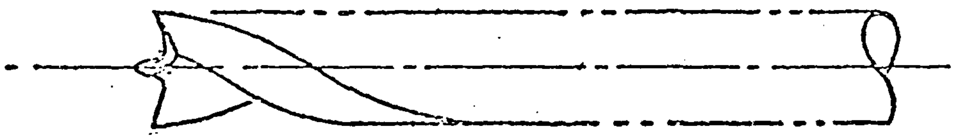


FIGURE B-7
SPECIAL DRILL POINT FOR
PRD-49 LAMINATES

FIGURE B-8
STEPPED DOUBLE MARGIN DRILL

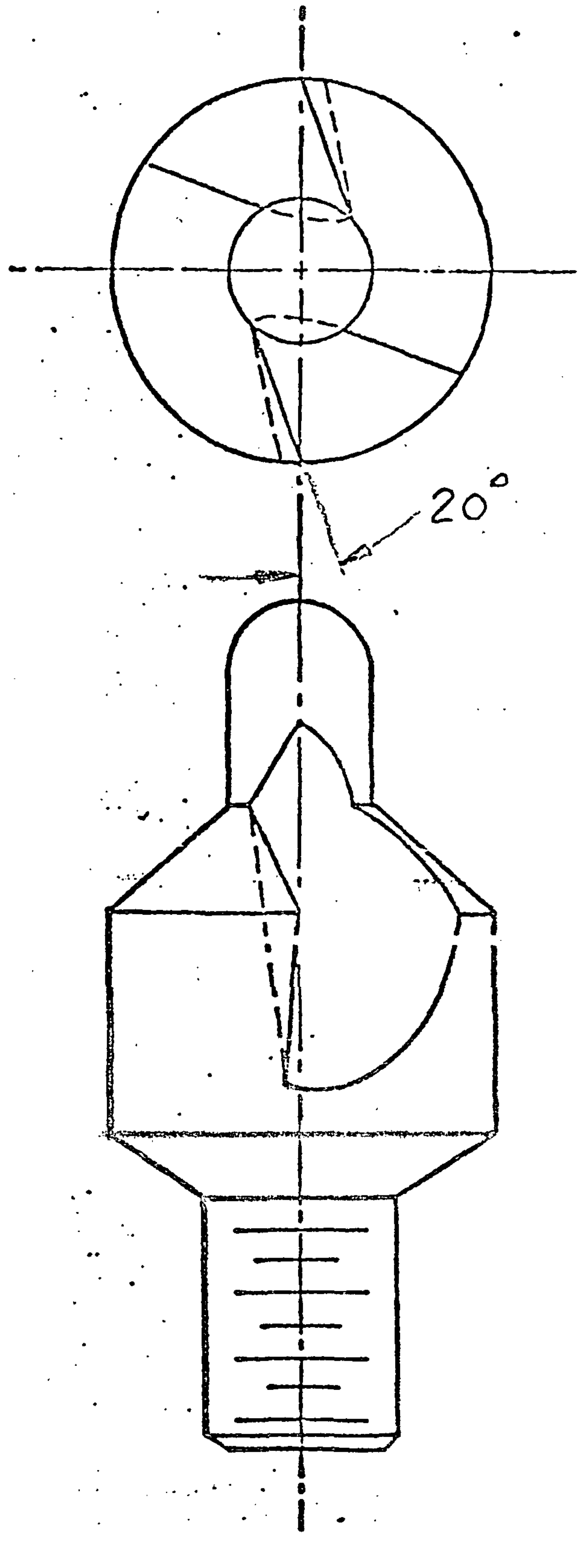


Figure B-9. Countersink A 4.76 mm Dia. (0.187 inch)

A second configuration was produced by reworking a standard 100° steel countersink as shown in Figure B-10. This tool was used with a standard countersink stop cage. All of the countersinking operations at Lockheed were performed using the modified tool shown in Figure B-10, and there was no evidence of dulling of the tool. Countersink or drill life should not be a significant factor in the cost differential involving PRD-49/epoxy, however, many more holes must be prepared before a definite determination of tool life can be made.

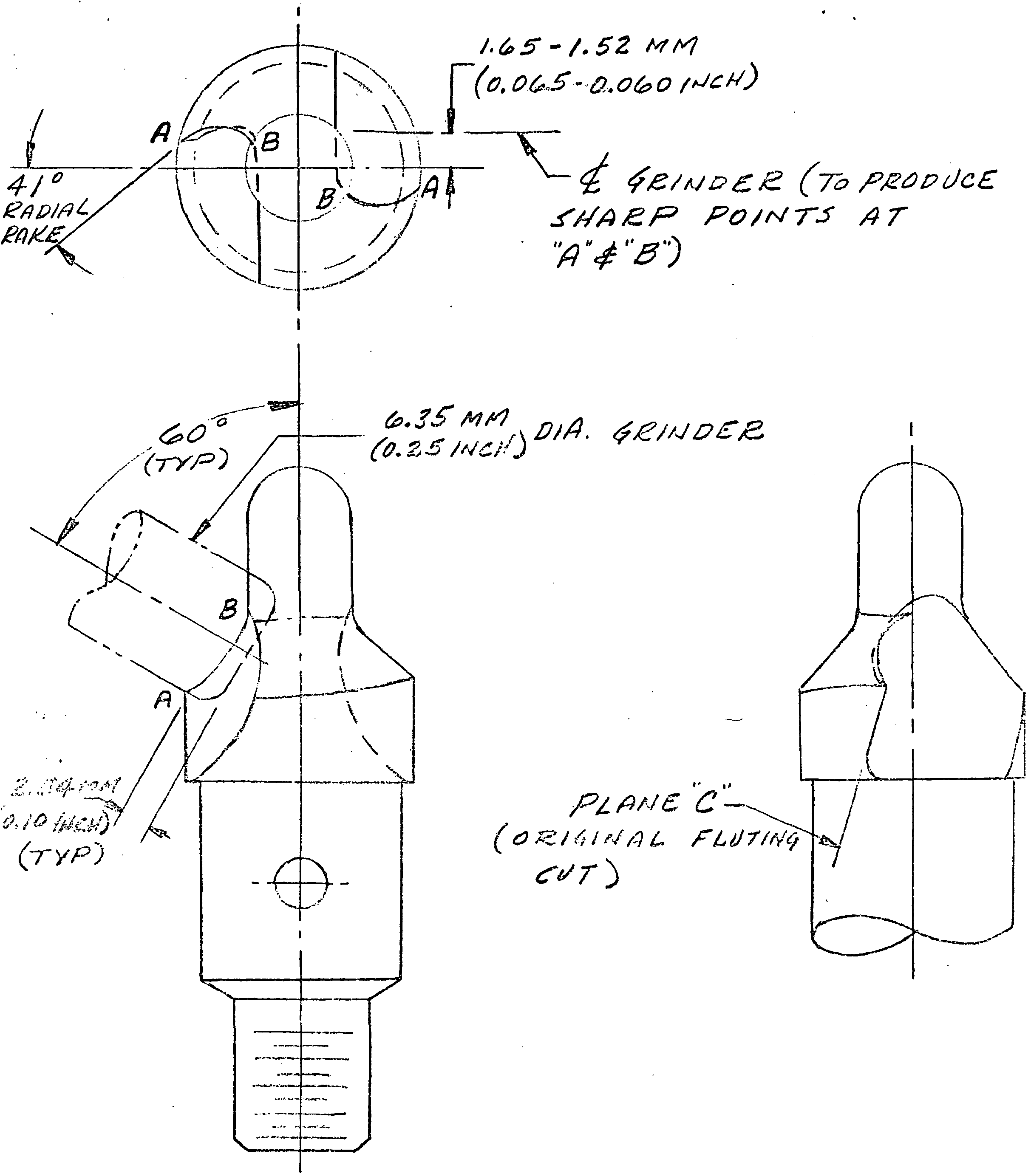


Figure B-10. Countersink B 4.76 mm Dia. (0.187 inch)